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THESIS

SATELLITE ON-ORBIT REFUELING: A COST EFFECTIVENESS ANALYSIS

by

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September, 1996

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SATELLITE ON-ORBIT REFUELING: A COST EFFECTIVENESS ANALYSIS

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ABSTRACT

With the ever-shrinking military budget constraints facing military and civilian contractors, the ability to extend the operational life of any system for minimal cost compared to a replacement is desirable. This fact has never been more true than in today's space industry. This thesis addresses the possibility of extending satellite life through the use of on-orbit refueling. Through compilation and analysis of satellite operational life span data, it is shown that maneuvering fuel depletion has a significant impact on satellite operations in geosynchronous orbit. If these satellites could be refueled economically this would prove not only cost-effective but also improve satellite tactical employment for space support to the warfighter. Through the manipulation of satellite data, launch/design cost, on-orbit refueling vehicle design/construction costs and on-orbit operational requirements, it can be shown that on-orbit refueling can be done cost effectively. Single versus multiple satellite refueling operations were evaluated to determine the concept's viability.

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I. INTRODUCTION

A. WHY SATELLITE REFUELING?

With ever-shrinking budget constraints facing the military and civilian contractors, the ability to get the most for your dollar has become a major factor in all programs. The ability to extend the operational life of any system, for minimal cost when compared to a complete system replacement, would obviously make that system much more desirable as well as salable. This fact has never been more true than in today's space industry. Budget constraints have become the driving force, resulting in the continual search for more efficiency and flexibility from new satellites and systems.

If a satellite design life of five years could be extended to seven years, over a twenty year period a savings of more than one satellite and its associated launch cost could be realized. Currently, in the opinion of many, fuel usage drives the operational restrictions placed on satellites, with fuel being closely managed in order to utilize the complete satellite design life. A fully functional satellite, which has depleted its maneuvering/station keeping fuel to reserve levels, is no longer usable. The reserve fuel must be used to boost the satellite into a super-synchronous orbit in order to vacate the geo-synchronous slot for a replacement satellite. A second option is to use all fuel for on station maneuvering and allow the satellite to drift toward the nearest "dead point" which further complicates the space debris problem. If the satellite actual life exceeds design life, the operational limiting factor could very well be onboard maneuvering fuel. The ability to replenish satellite maneuvering fuel on-orbit could result in a significant satellite operational life extension. With little or no fuel budgeted for contingency

operations, the need to maneuver a communications or intelligence satellite to cover evolving regional conflicts (such as Iraq/Kuwait or China/Taiwan) could directly and disastrously affect the initial design fuel budget. Once a satellite/fuel load is placed on orbit, tactical repositioning would directly impact satellite life expectancy through the diversion of fuel budgeted for normal station keeping operations to satellite maneuvers/repositioning. Fuel considerations/limitations may also preclude development of operational concepts necessary to meet lower priority tasking/requirements. This "husbanding" of limited onboard fuel assets for strategic missions could negatively impact "space support to the war fighter". Again, the ability to replenish satellite maneuvering fuel on-orbit could result in significant operational flexibility, as well as increased operational life expectancy.

There have been numerous proposals to develop a "satellite launch on command" capability to cover regional conflicts such as those previously mentioned. During operation Desert Storm there were grossly insufficient regional communications channels/capabilities, with "FLASH" message delivery often taking several days. The deployment of on-orbit units such as a "duty" communications or intelligence satellite which could be maneuvered to cover the latest global "hotspot" could compensate for this shortfall. However, the satellite must be able to maneuver freely without concern for onboard fuel. This concept would be feasible if a satellite was refuelable "on-orbit".

The Westar 6 communications satellite, launched 4 February, 1984, suffered a PAM-D upper stage booster malfunction, stranding the satellite in a useless orbit. The apogee kick motor and onboard thrusters were used to boost the satellite to a higher orbit

for eventual retrieval by STS 51A. Total rescue cost exceeded $10.5 million. On 4 April, 1983, the launch of TDRS 1 experienced a second stage IUS malfunction, which required the use of 30 onboard thruster burns, consuming 370kg of maneuvering fuel, to obtain desired orbital positioning. Most recently, a GPS Block III satellite experienced booster malfunction, which left the satellite in an orbit too low to fulfill mission requirements. The onboard fuel was sufficient to boost the satellite to the required operational orbit, however, the maneuver would consume the fuel budget for the entire satellite life span. The satellite was thus necessarily boosted into super-sychnronous orbit, fully functional but operationally worthless. Had these satellites been on-orbit refueling (OOR) capable, a refueling mission after initial satellite altitude repositioning could have restored the maneuvering fuel reserves and saved much of the cost of the replacement satellite/ associated launch or retrieval efforts. [Ref 7,Comsats]

B. TACTICAL APPLICATIONS

As airborne refueling revolutionized tactical and strategic aviation, an on-orbit satellite refueling capability could result in a similar expansion in mission scope and flexibility in space. The OOR capability would allow operational necessity vice fuel considerations to drive mission tasking. No longer must each satellite repositioning be weighed against the tactical or strategic "benefit" which often falls short when considering a limited maneuvering fuel budget. [Ref 1,p.9-14]

An OOR capability could allow for an actual decrease in initial onboard fuel budget, thus allowing for increased payload/mission capability. Current satellite design requires an onboard fuel capability sufficient to meet the design life expectancy.

However, care must be taken to not oversupply onboard fuel, to preclude the satellite reaching its end of life with several hundred pounds of now useless on-board fuel. With launch costs up to $10,000/pound to geosynchronous orbit (about 35,000km), elimination of excess onboard weight is critical. Engineers must also consider the fact that many satellites exceed their scheduled design life, and hence may require additional on-board fuel if this proves to be the case. Engineers and designers must carefully balance all these factors and then hope for the best. I can think of nothing more frustrating than being forced to discard a fully functional satellite due to station-keeping fuel depletion. Although you may be gambling on a successful refueling mission, if the initial fuel budget is sufficient to meet design life, should actual satellite life exceed design life, OOR capability could solve the initial design dilemma. The tradeoffs would involve the actual weight of the docking/refueling apparatus versus the launch cost/weight penalty. However, if the weight increase would not dictate a shift to a larger payload capable launch vehicle, the impact would be minimal.

The scope of this evaluation will be limited primarily to satellites in geosynchronous orbits (GEO). Due to the associated system redundancy required for the safety of manned expeditions and the associated expense, this evaluation will be limited to unmanned vehicles. Specific refueling vehicle design will not be addressed. The points which must first be addressed are:

- Is on-orbit refueling (OOR) technologically feasible?
- Is fuel actually a limiting factor in GEO satellite operations?
- Is OOR cost effective?

II: SATELLITE ON-ORBIT SERVICING

A. BACKGROUND

On-orbit satellite servicing is not a new idea. The concept was recently explored in 1984, when NASA first discussed use of the space shuttle to retrieve, refuel and repair imaging reconnaissance satellites in order to extend their operational life spans. [Ref 2] This concept was first successfully demonstrated in April, 1984 during the recovery and repair of the Solar Maximum satellite. This shuttle mission was the first to use a direct insertion technique, which resulted in a shuttle apogee of 250nm, necessary to reach the 265nm altitude of Solar Max.[Ref 3, p.42-44] The successful rendezvous with the satellite allowed astronauts, using extra-vehicular activity (EVA) suits and the shuttle manipulator arm, to successfully retrieve the 4,500lb satellite (after one initial failure) for repair in the shuttle bay. Replacing a failed General Electric attitude control box, the coronagraph's main electronic control box, and installing a vent port baffle to prevent plasma entry into satellite electronics took the two astronauts approximately six hours. The repair was made possible due to the Goddard/Fairchild multi-mission spacecraft modular design employed on Solar Max. [Ref 4, p.18]

NASA's success with Solar Max led to the scheduled on-orbit attempt by shuttle mission 51-I to repair the $85-million Hughes/Navy Leasat 3 satellite. [Ref 5,p.48] At an altitude of 242nm, the 7.5ton Leasat 3 failed to activate after its initial deployment on April 12, 1985 by STS-51D. A previous attempt by the mission 51-D crew to deploy the manual arming lever, using the shuttle manipulator arm, was unsuccessful. After some initial difficulty in retrieval, the satellite's sequencer was disabled and the booster motor

safety pinned. Two small panels were removed and a spin bypass unit was installed to allow Leasat 3 to process commands directly from the ground. After connecting a battery powered control box, the satellite's 7.5ft omni antenna was deployed, which concluded the initial EVA at 7.62 hours. The second EVA of 2.45 hours consisted of the installation of temperature sensors on the motor nozzles, removal of previously installed safety pins and the activation of two 13 hour timers which precluded the processing of ground signals for 13 hours, in order to allow for safe withdrawal by the shuttle prior to satellite activation. [Ref 6, p.21-23] NASA received $8.5 million for conducting the successful repair effort. Compared to the initial satellite cost of $85 million (plus associated launch costs) coupled with a replacement satellite/launch costs, the repair was truly a bargain. [Ref 7, Comsats]

The most recent and probably most famous instance of on-orbit servicing was conducted by STS-61, to repair the $1.5 billion Hubble Space Telescope (HST). After initial launch in 1990, scientist discovered the HST had several problems, the most significant being the inability to focus (due to improperly ground optics) as well as a "jitter" problem related to the solar arrays. The very rapid temperature change during day/night transitions resulted in array deflections, which although extremely minute, directly impacted HST operations. The original arrays were replaced with a shielded, 9 coil spring mounting array, with an onboard braking control to eliminate solar induced array movement. Servicing also included: the installation of corrective optics space telescope axial replacement (Costar) to correct HST's vision flaws, swapping a second-generation wide field camera, replacement of a failed relay box in the Goddard High

Resolution Spectrograph (GHRS), installation of a coprocessor module to add computer memory, replacement of three failed gyroscopic units, and change out of the magnetometer. [Ref 8, p.28-29] Although the HST was designed with multiple replaceable parts for periodic on orbit repair, many of the scheduled repair operations involved units or access panels which were not designed for on-orbit servicing. [Ref 9, p.14-16] Servicing efforts proved a resounding success, further justifying the on-orbit servicing satellite design concept.

The concept of on-orbit refueling was successfully demonstrated on shuttle mission 41-G, by CDR David Leestma and Kathryn Sullivan. This proof of concept, using the Orbital Refueling System (ORS), demonstrated the capability to refuel satellites currently on-orbit which have not been specifically modified for refueling operations. This process involved special penetration of the fueling system. [Ref 9, p.15]

Although the use of shuttle manned EVA evolutions to conduct on-orbit servicing has proven successful in LEO, shuttle operational limitations preclude such operations above 400nm. [Ref 10, p. 179] Satellites which operate in MEO or GEO with typical altitudes of as high as 22,000nm are not accessible to shuttle flights at this time. However, as successful as NASA has been in conducting on-orbit satellite repairs, the presence of manned evolutions significantly increases the cost. However, modular replacement or refueling operations using unmanned vehicles requires the satellite to be designed with this eventuality in mind. Several on-orbit service vehicle (OSV) design options have been evaluated, with the most significant being the Orbital Maneuvering

Vehicle (OMV), designed for NASA by the TRW Space and Support Group. Its primary missions include:

- Spacecraft retrieval, reboost, deboost or viewing
- Spacecraft on-orbit servicing, including refueling and component replacement
- Space station construction and logistics support
- Large observatory service (HST) from either space station or shuttle
- Experiment carrier for sub-satellite missions [Ref 1, p.32-33]

NASA plans call for the OMV to be deployed via the space shuttle and later retrieved for return to earth for periodic servicing. The OMV is 15 feet in diameter and is 56 inches in length (see Figures 1, 2 and 3). It incorporates a fully modular configuration which allows on-orbit replenishment of fuels as well as replacement of modular units (ORUs). The OMV was designed to service satellites in LEO, polar orbits (inclinations above 57 degrees) which are not accessible by shuttle operations. [Ref 11,p.29-33]

Although intended for use in LEO operations, the OMV unmanned servicing vehicle concept can be applied to satellite refueling operations in GEO. However, automation maneuvers must be precise and assured. The two major limiting factors of on-orbit refueling are satellite rendezvous/docking and fuel (fluid) transfers.

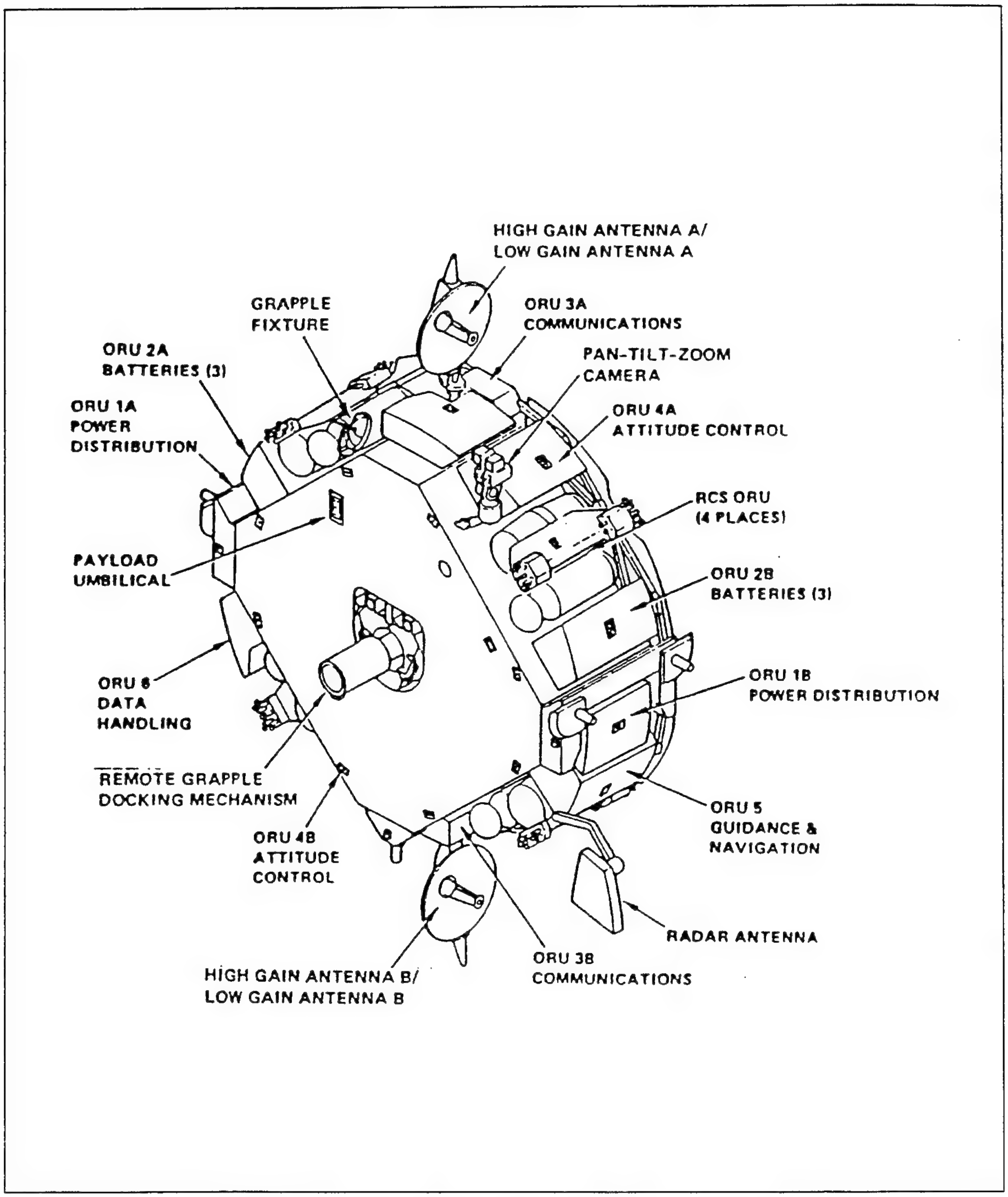


Figure 1. OMV Service side for ORU replacement.

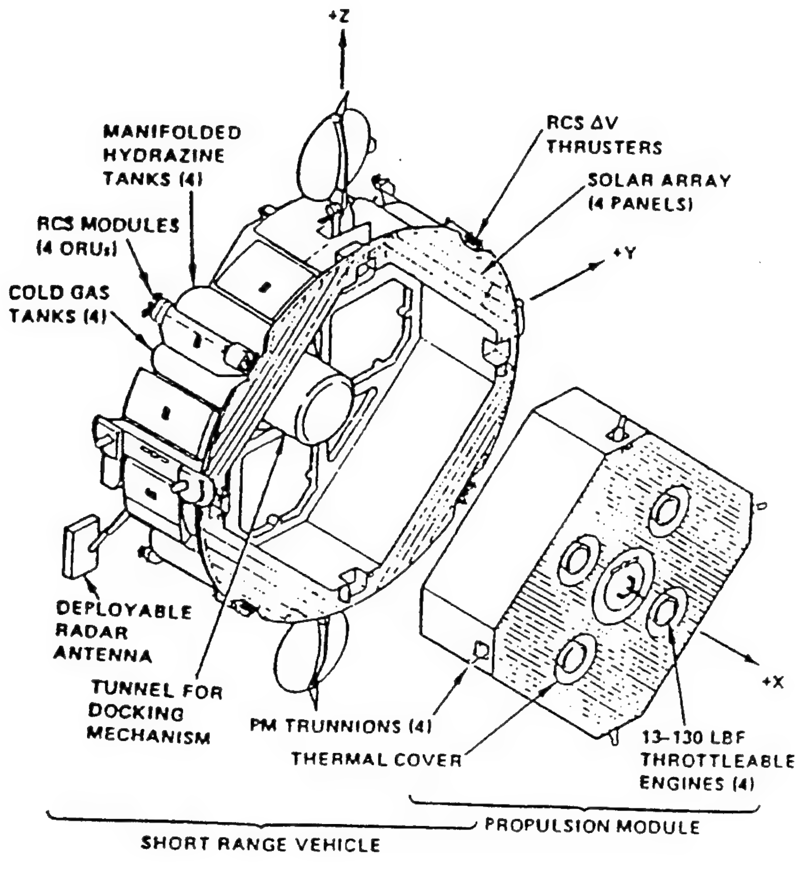


Figure 2. OMV propulsion side - showing replaceable propulsion module

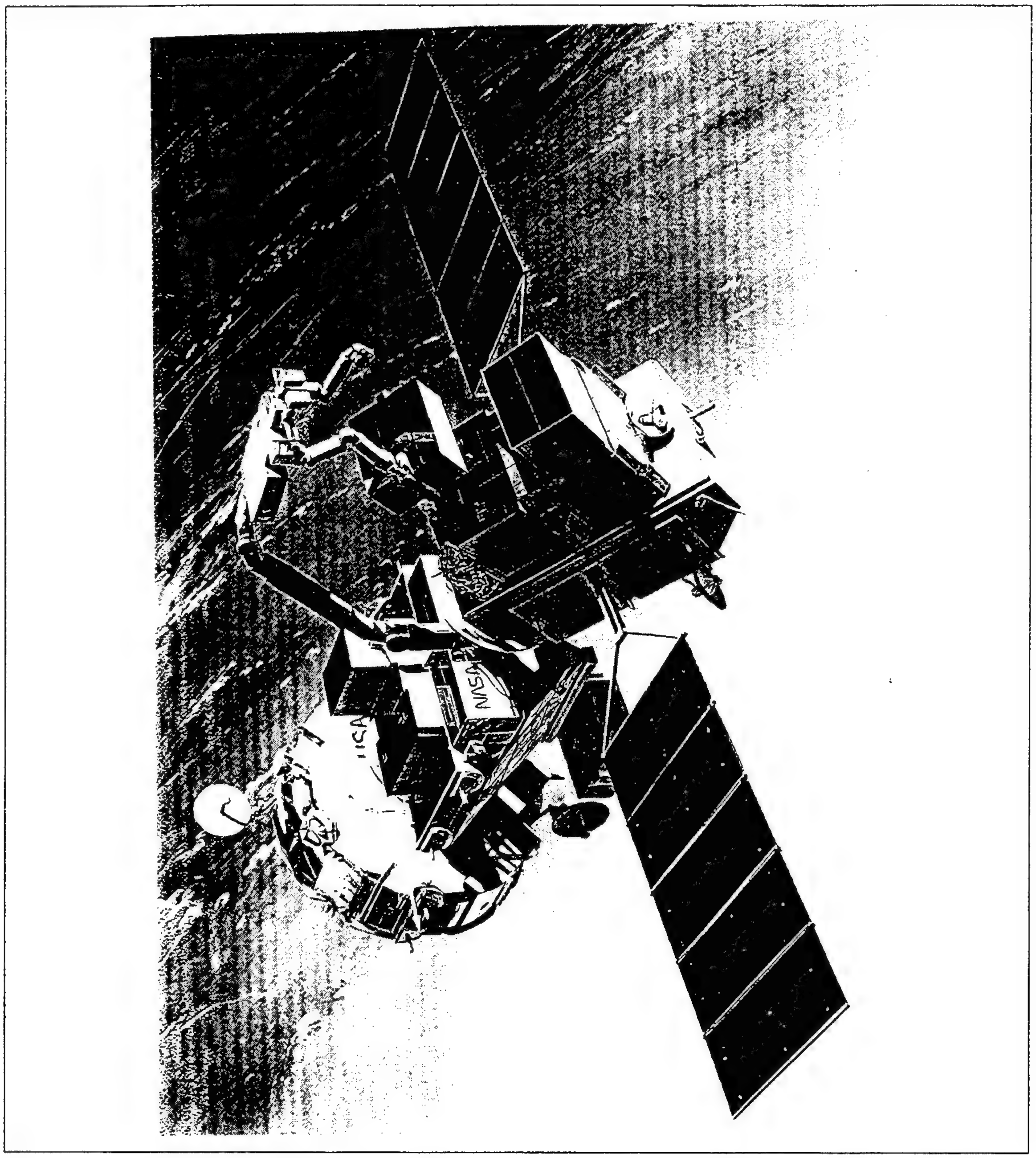


Figure 3. OMV shown docked with satellite.

A semi-autonomous navigation and docking capability has been developed. Using optical reference patterns and a computer vision system to determine relative position and attitude, semi-autonomous docking has been evaluated using both

“passive”and “active” targets. Active targets have visual reference “cuing markers” installed prior to launch, while passive targets do not. Only the primary spacecraft is under active control, with the docking target completely independent. Cuing markers consist of geometric patterns which provide orientation and/or distance reference using the geometric patterns (see Figure 4). Nesting a series of these optical patterns provides a means by which an autonomous cross-correlator guided craft can determine its range and orientation during approach and docking maneuvers (see Figure 5). At the furthest distance the largest pattern is used as a reference. Upon closer approach, the correlation pattern grows larger and larger in the field of view until it actually reaches a range where the complete target is no longer visible. At this point a smaller nested array, located at the center of the first pattern is discernable, and the system begins to process the second pattern for range and orientation data. The simple task of recognizing a single visual pattern, in a cluttered environment is well within the capability of an optical cross correlator. This single-function vision device can accurately provide the necessary recognition and spatial orientation necessary for semi-autonomous navigation, landing and docking in three dimensional space, without natural landmarks. The single-functional optical cross- correlator, using video input from a simple imaging camera and optical correlation-plane output coupled with standard star tracker software, provides sufficient information for spacecraft navigation and docking maneuvers in space. [Ref 11, p.5049-55]

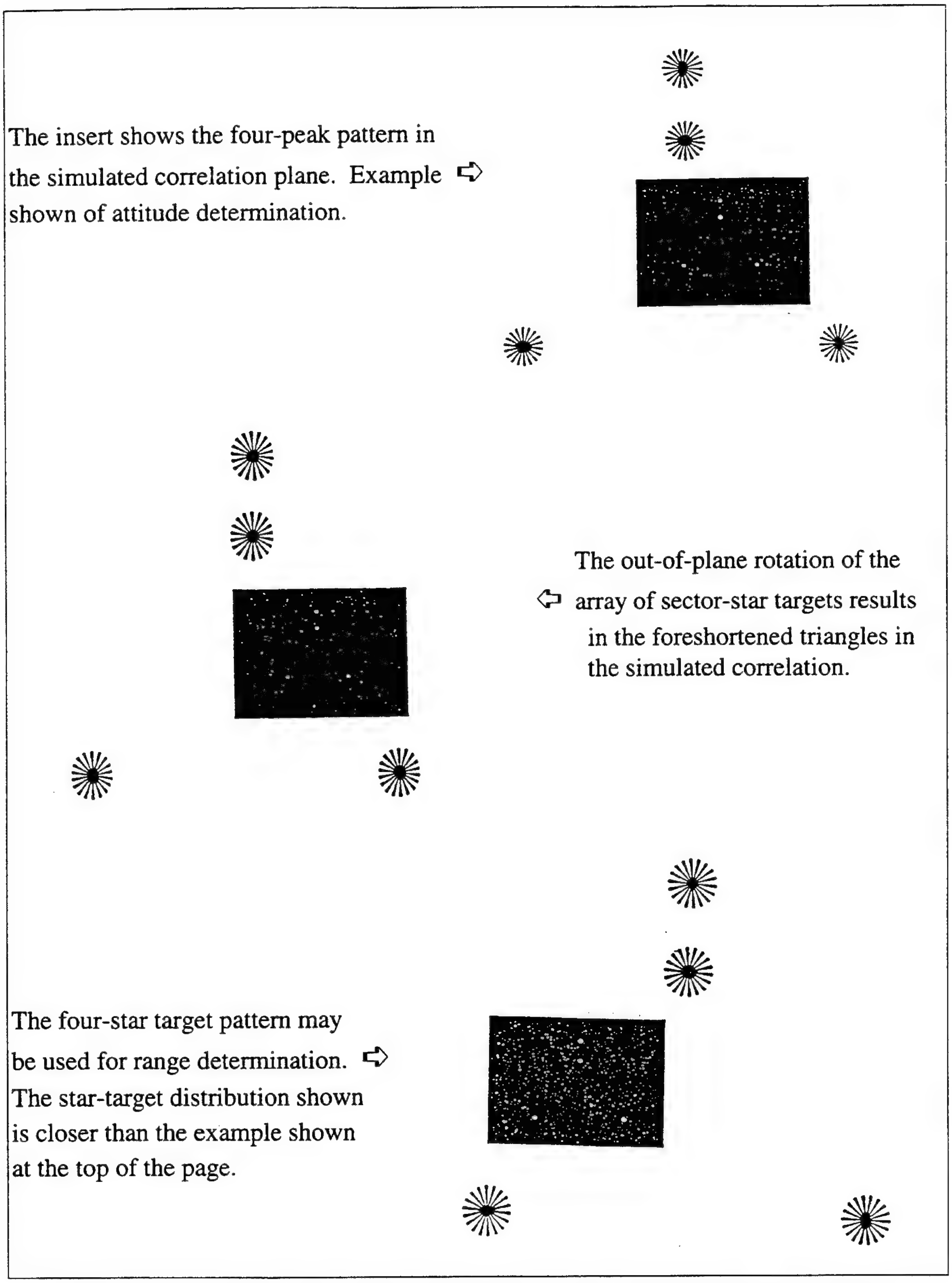


Figure 4. Four-sector star target visual patterns used for attitude determination.

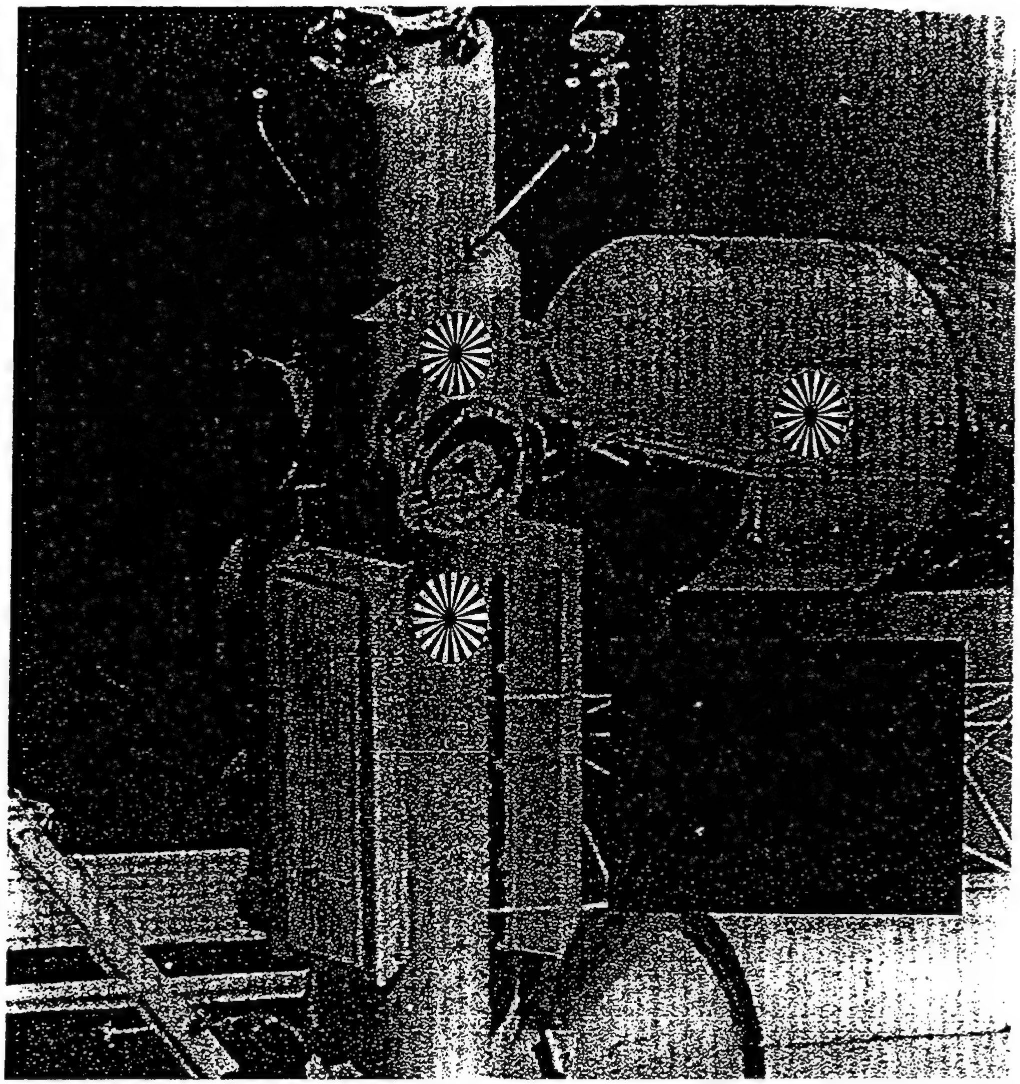


Figure 5. Simulated docking port with three sector-star targets attached.

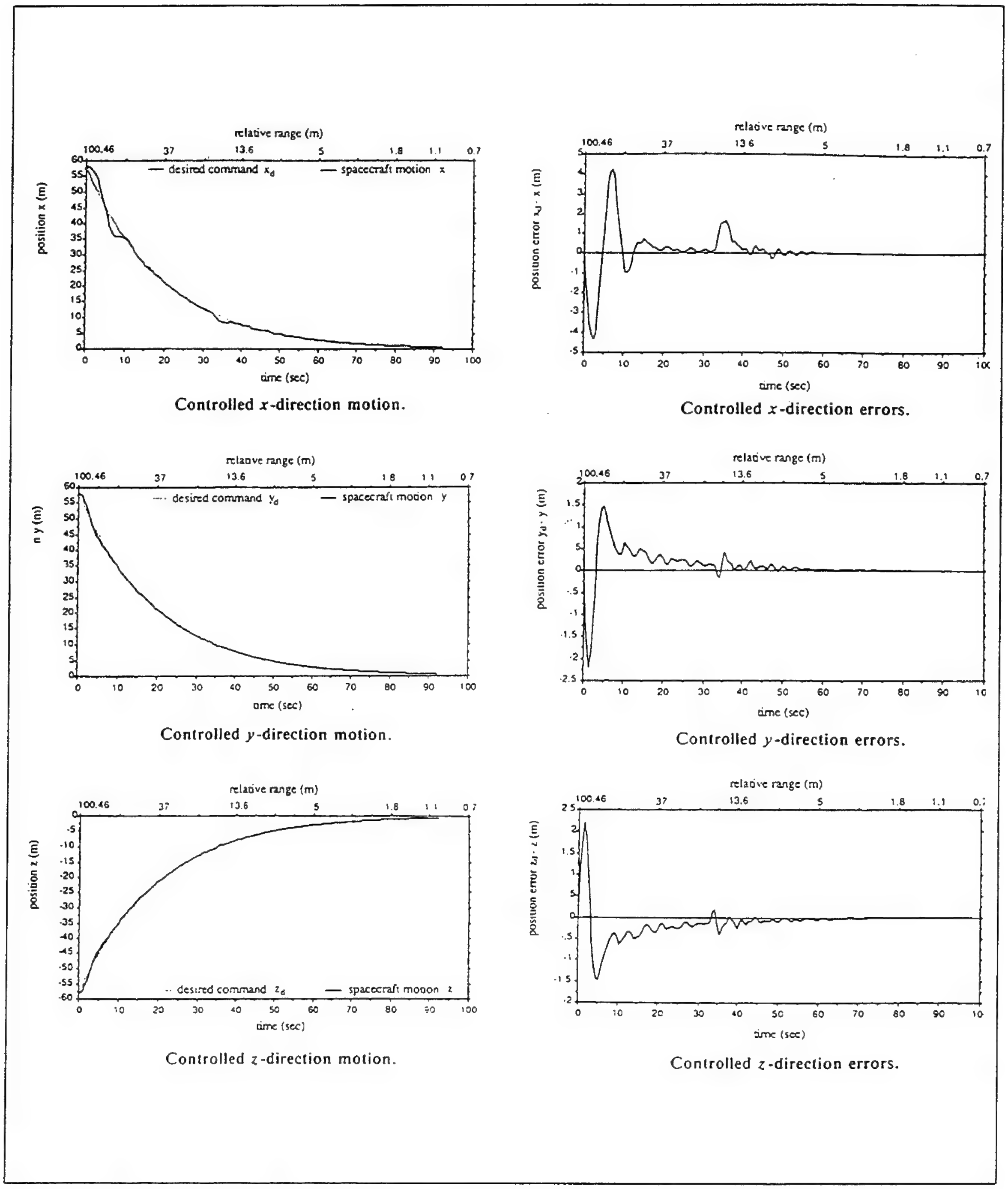
The following conditions are required:

- The spacecraft must be a rigid body with three pairs of gas jet thrusters mounted along the principle axes for control of spacecraft translational and rotational motion.
- A pinhole camera is rigidly mounted on the spacecraft. The cuing marks located on the docking target platform are always visible to the spacecraft.

Computer-vision based methods have several advantages over the use of sensors such as laser, infrared, radar, GPS or INS. Specifically, the estimation accuracy of the relative position and vehicle orientation improves as the range between the two vehicles decreases. Thus, the control accuracy of the system control loops improves proportionally (see Figure 6 and 7). Hence, the computer-vision based control and docking system is well suited for precise maneuvers required for autonomous satellite docking. [Ref 12, p.649]

Currently, Russia conducts resupply missions to the MIR space station using the Progress spacecraft, which employs the Kurs automatic rendezvous and docking system. A back-up remote control docking capability has been developed, which although not autonomous, does not require manned participation on-orbit. The TV-aided system enables ground-based controllers to remotely fly the spacecraft for rendezvous and docking. A television camera provides live images to the ground-based cosmonaut, who will dock the spacecraft using two control sticks, much as if he were actually onboard. A successful demonstration of this system was conducted in 1993 by a cosmonaut onboard the MIR space station. [Ref 13,p.70] Hence, docking unmanned refueling missions should not pose a technical problem.

There are three major methods of on-orbit propellant transfer: direct fluid transfer, tank to vehicle transfer, and propulsion module to vehicle transfer. Direct fluid transfer, as implied by the name, involves the transfer of fuel from the servicing vehicle directly to the satellite tank. Tank to vehicle transfer involves the transfer of full fuel tanks to the satellite as orbital replacement units (ORU). Propulsion module transfer involves the



Controlled x-direction motion.

Controlled x-direction errors.

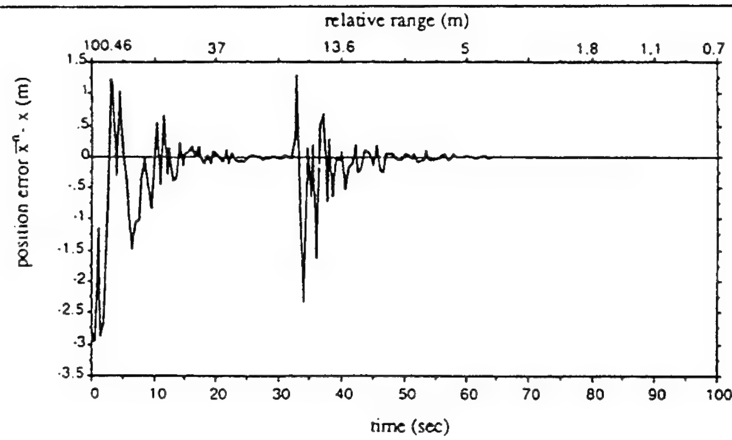
Controlled y-direction motion.

Controlled y-direction errors.

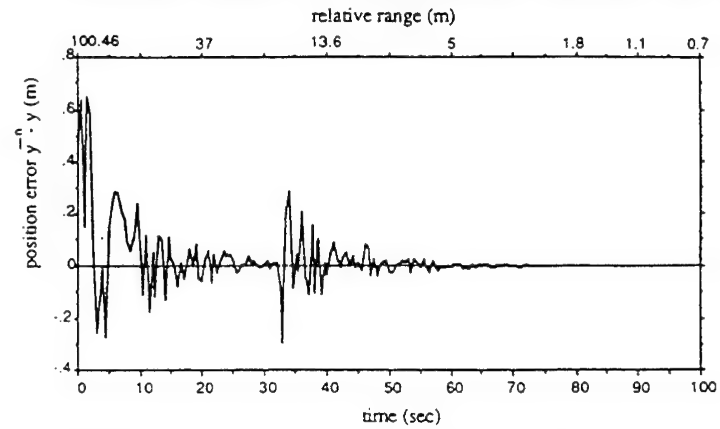
Controlled z-direction motion.

Controlled z-direction errors.

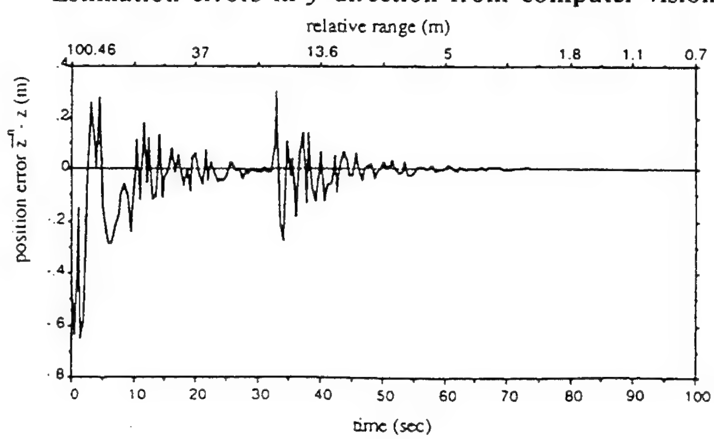
Figure 6. Graphs show correlation to controlled X, Y, and Z axis motion and associated error as range decreases.



Estimation errors in x direction from computer vision sys-



Estimation errors in y direction from computer vision sys-



Estimation errors in z direction from computer vision sys-

Figure 7. Depicts system estimation errors in X, Y, and Z directions as range decreases.

installation of a complete propulsion system to include the propellant tanks, main engines, thrusters, fluid lines and management systems and controls. Trade studies and evaluations revealed that the direct fuel transfer was the most cost effective and feasible technique, and has the least impact on spacecraft design. Additionally, this method allows full utilization of onboard fuel, while requiring the smallest number of interfaces. Disadvantages involve safety concerns related to actual displacement of propellant from one tank to another. The primary focus of this study involved fueling in space for long duration missions, such as manned Mars exploration. Hence, concerns of lengthy fuel transfer durations and the associated large pumping capability required would not apply to satellite refueling, due to the much smaller relative fuel quantities required for satellite stationkeeping/maneuvering operations. [Ref 14, p.1423-33]

The transfer of fuel is complicated by many factors, the most significant involving a means of pumping fuel in a near weightless environment and the necessity to vent waste gases from the receiving tank as it fills, without venting fluids. Fuel cannot be gravity fed for obvious reason. The most promising on-orbit servicing method for direct fuel transfer under these conditions involves the use of a screen-channel liquid acquisition device (LAD). Designs for screen LADs are usually conduits, with walls made of porous, fine mesh screen, which are routed around the tank perimeter and manifold at the tank outlet. [Ref 15, p.1099-1106]

The capability for fuel/fluid transfer on-orbit was successfully demonstrated on shuttle mission 41-G during astronaut EVAs. However, EVAs are an expensive option and do not meet stated goals of autonomous operations. NASA conducted Fluid

Acquisition and Resupply Experiments (FARE) in shuttle middeck experiments to demonstrate LAD techniques for transferring liquids in zero gravity (see Figure 8, 9 and 10). The first experiment occurred on STS 53, launched December 2, 1992, with the second on STS 57 in June of 1993. The objective was to demonstrate tank refilling, low gravity propellant center of gravity control, and expulsion efficiency. A fluid expulsion

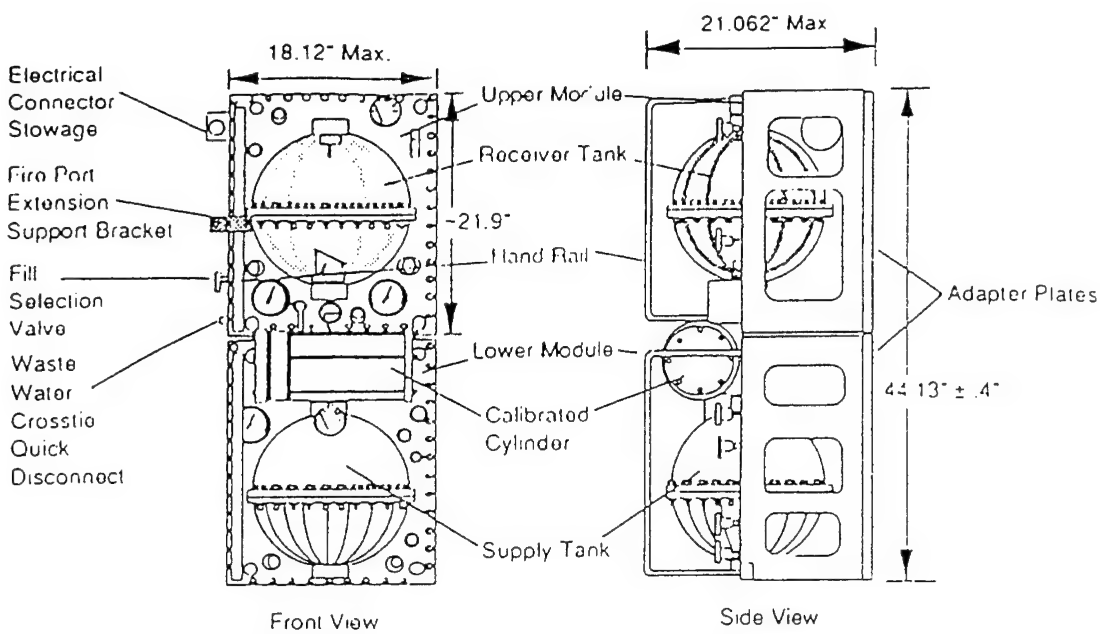


Figure 8. FARE test configuration in shuttle bay.

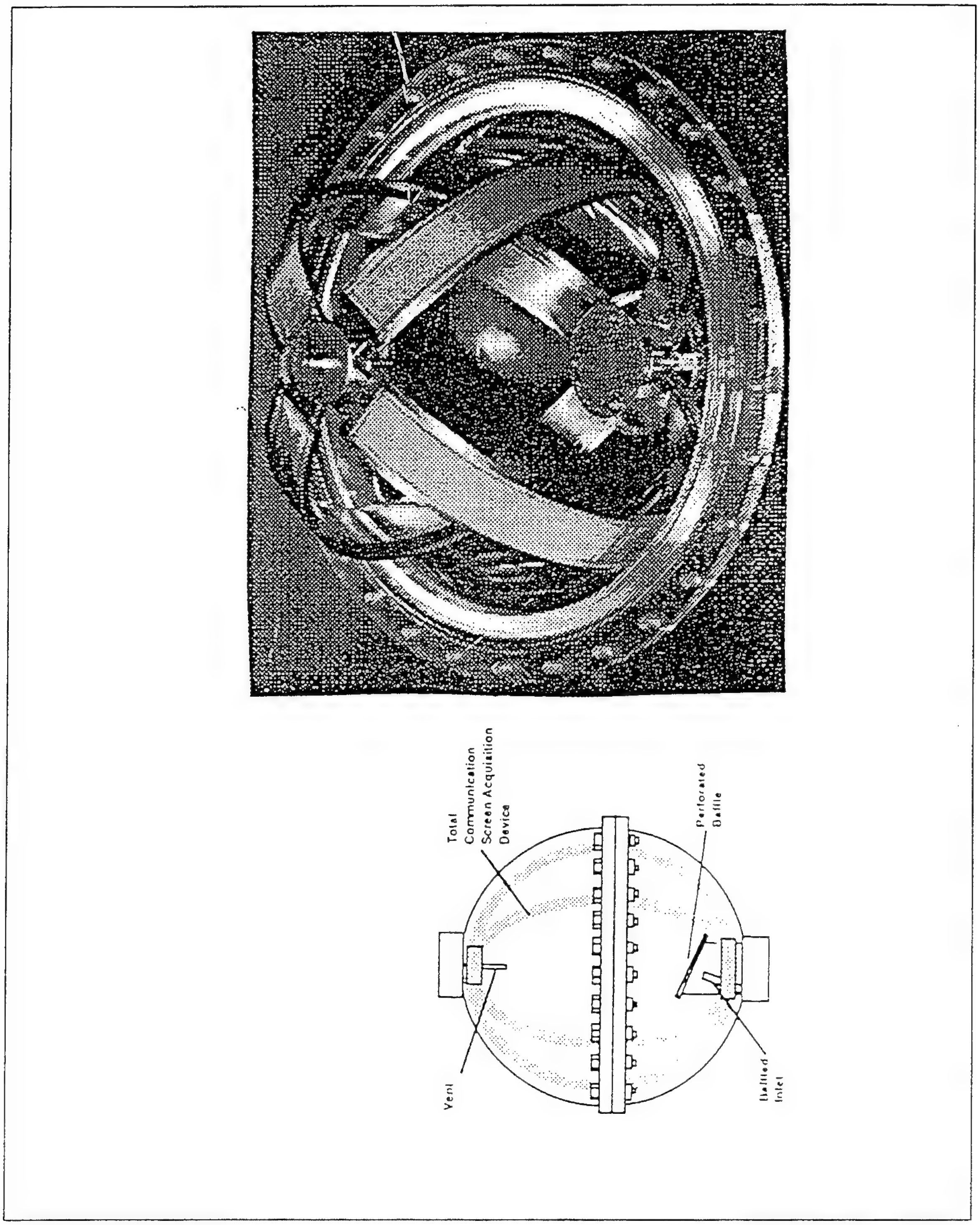


Figure 9. FARE 1 receiver tank configuration.

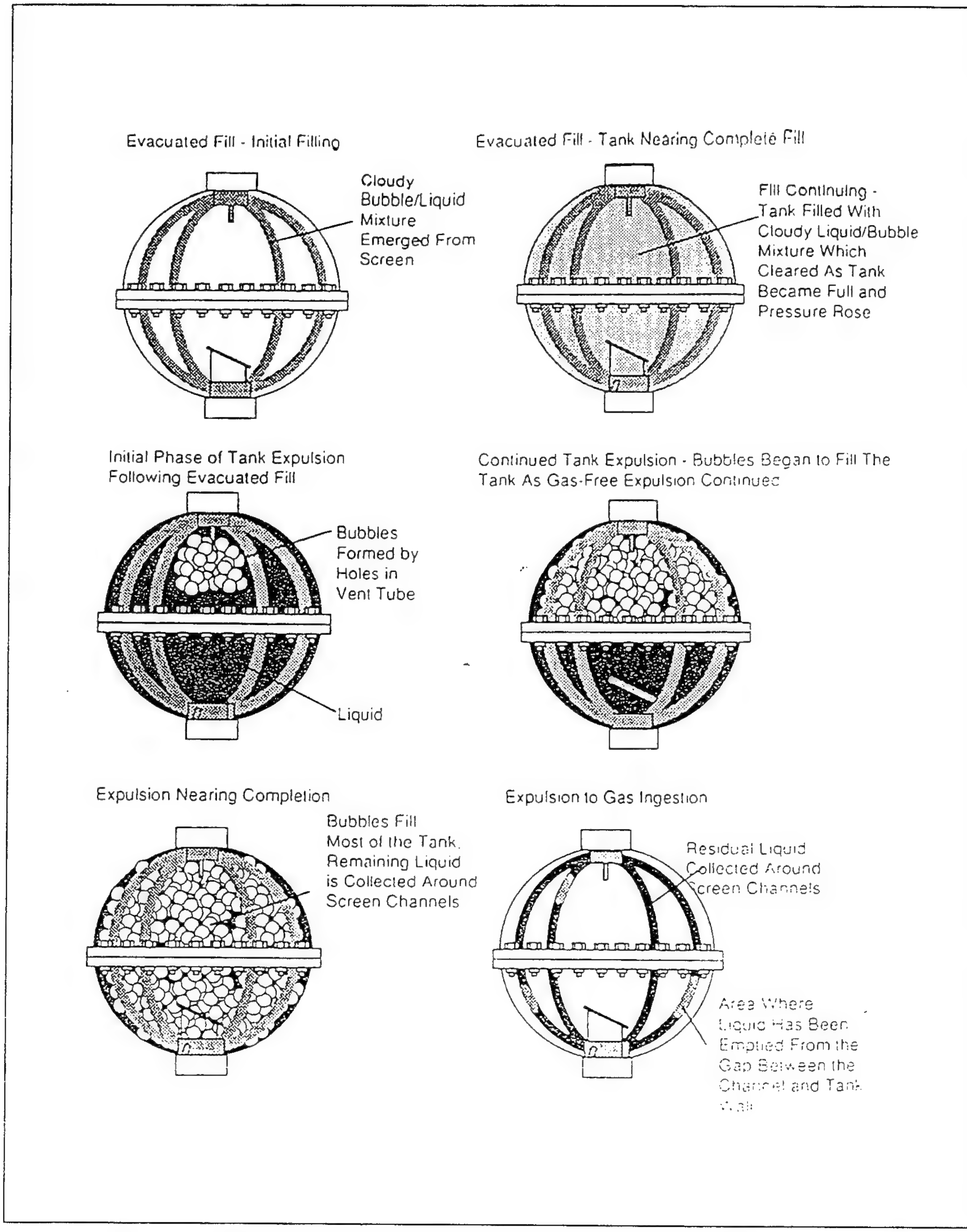


Figure 10. Evacuated Fill/Expulsion Test Results.

efficiency of at least 98% was obtained and final fill levels of greater than 95% were routinely achieved without the venting of liquid overboard, thus validating capabilities to refuel spacecraft on-orbit. [Ref 16, p.1-12]

Although the problems presented by autonomous docking and on-orbit fluid transfer are technoligically challenging, they remain well within the range of current mechanical and scientific capabilities. However, production and integration costs may dictate actual system applications in order to assure maximum cost effectiveness. The most obvious question at this juncture remains, does satellite maneuvering fuel actually impact satellite life span and operations to the point that on-orbit refueling is necessary?

III. SATELLITE DATA ANALYSIS

A. SATELLITE LIFE-SPAN ANALYSIS

Prior to conducting a cost effectiveness analysis of on-orbit satellite refueling, the data concerning a satellite's actual life and its design life must be examined. Without addressing the tactical maneuvering of satellites and associated fuel considerations, satellite life expectancy is an important factor in determining if the need to refuel exists. The satellite data listed in Appendix A - Part 1 (page 1 and 2), was compiled from Jane's Spacecraft 1984-1996, Interavia Space Directory 1986-1996, as well as inputs from various contractors such as TRW, Hughes, etc. The data set consists of U.S. satellites, launched to geosynchronous orbit, in the last 20 years. Analysis of satellites which have reached geosynchronous orbit and full operational status (i.e., they have not experienced launch-related failures) follows:

Sample Size	Mean Design Life	Std Dev	Mean Actual Life	Std Dev
57 Satellites	8.11 years	1.94 years	11.41 years	3.08 years

Table 3-1

With the satellite actual life exceeding design life by an average of 3.3 years, this data supports the hypothesis that satellites typically exceed design life expectations. However, these statistics are somewhat misleading, as 28 of the 57 subject satellites are still operational (SOPER). This computation assumes each satellite, even if still operational, has reached end-of-life (EOL). (GOES 5/6 experienced imagery failure (the primary mission), but continued service in a data relay mode until fuel depletion. The primary mission role was used in assessing mean satellite life.)

Evaluating only the satellites which have actually reached EOL shows:

Sample Size	Mean Design Life	Std Dev	Mean Actual Life	Std Dev
29 Satellites	7.66 years	2.09 years	10.43 years	3.56 years

Table 3-2

With the satellite actual life exceeding design life by an average of 2.77 years or 26.55 percent, this data is .6 years less than the total sample average life delta of 3.3 years, but within the standard deviation.

Evaluating only the SOPER satellites yields:

Sample Size	Mean Design Life	Std Dev	Mean Actual Life	Std Dev
28 Satellites	8.46 years	1.69 years	12.43 years	2.10 years

Table 3-3

Examination of the SOPER satellite subset reveals that satellite actual life currently exceeds design life by 3.97 years or 31.93 percent. The SOPER satellite subset life delta is greater than both the entire satellite sample delta (3.08 years) and the EOL satellite delta (2.77 years). As these satellites are still operational, each passing year will increase the life delta until all satellites have reached EOL. Even without considering these additional active years, there is significant evidence that satellite actual life consistently exceeds satellite design life, regardless of sample data chosen. However, for the purpose of this cost effectiveness study, the average satellite life delta of three years will be used.

B. SATELLITE FUEL ANALYSIS

The next question which must be addressed is, did fuel play a significant role in satellite failures? Fuel depletion is an obvious fuel impact, but other factors must be considered. Satellites which are still operational, but are nearing maneuvering fuel limits/depletion, often continue East/West stationkeeping but cease North/South station keeping in order to conserve maneuvering fuel. This practice results in geosynchronous satellites assuming inclined orbits, which impacts the satellite's area of coverage or "footprint" on the earth. This will impact coverage in peripheral areas at the northern and southern extremes of coverage. As the inclination increases (about 1 degree/year without correction) the affected area increases as well. Satellites conducting fuel conservation operations (FCO) are thus impacted by fuel limitations.

Examining the cause of failure for the 29 satellites which have reached EOL reveals that nine satellites (31 percent) ceased operation due to maneuvering fuel depletion. This number increases to 11 satellites (37.93 percent) if the fuel depletion of GOES 5/6 is considered. Examining the SOPER satellites reveals that 19 of 28 (67.85 percent) satellites are currently conducting fuel conservation operations. Comparing the EOL and SOPER data:

	$\bar{x}$ Design Life	$\bar{x}$ Actual Life	Life Delta Δ	Δ%	Fuel Impact
EOL	7.66 years	10.43 years	2.77 years	26.55%	37.93 %
SOPER	8.46 years	12.43 years	3.97 years	31.21%	67.85 %

Table 3-4

Even without the inevitable increase in the SOPER satellite life delta, with current satellite design life ranging from 10 to 15 years (see Appendix A, p. 3-5), fuel considerations in the future will continue to significantly impact satellite operations.

A projection for the satellites launched since 1990 shows:

	$\bar{X}$ Design Life	$\bar{X}$ Actual Life	Life Delta Δ	Δ%	Assumption
Since 1990	11.37 years	14.89 years	3.52 years	31.21%	No Increase
Since 1990	11.37 years	15.45 years	4.09 years	35.87%	Projected

Table 3-5

Both "no increase" and "projected increase" in satellite design life versus actual life options are shown, with the projected increase based on the EOL/SOPER satellite design/observed life data. Although this projection is rather crude, even using the current satellite life delta of 31.21 percent, this data indicates that fuel considerations are becoming increasingly more significant.

Combining the entire satellite sample and associated fuel considerations, 30 of 57 (52.63%) satellites experienced some fuel-related operational impacts, with 20 percent failing due to fuel depletion. A convincing argument can be made that fuel limitations have a significant impact on satellite operations and that an on-orbit refueling capability could play a major role in solving this problem.

There are many alternative solutions to the satellite fuel problem other than on-orbit refueling. Many satellites, such as INTELSAT 706, are carrying additional fuel to preclude a fuel depletion problem. However, there are satellites, such as GALAXY 5/7, which do not carry sufficient fuel to meet expected design life. UFO satellites, which

previously experienced a fuel surplus, have recently seen this benefit eliminated due to a payload-for-fuel substitution. Additional fuel for UFO is not an option as the satellite is currently within 50 pounds of maxing out the launch vehicle payload capability. [Ref 17] As satellite design life continues to increase, there must be a point where it becomes economically and physically impossible to provide sufficient onboard fuel to meet design life. However, if tactical maneuvering of satellites is considered a viable mission requirement, on-orbit refueling is the only logical solution. The next question is, can it be done cost effectively?

IV: OOR DESIGN

A. OOR SPECIFICATIONS

As this is primarily a cost effectiveness analysis of the OOR concept, the specific design of the OOR will not be addressed. However, some general concepts and design configurations must be identified. Simplicity of design should be incorporated whenever feasible, utilizing as much existing proven space technology as possible. The OOR must have a configuration which would support launch on Titan IV (IUS) or comparable launch vehicle. Using the large payload fairing limits of the Titan IV, the OOR can be a maximum width of 5.1 meters and height of 15-26 meters in the stowed configuration. Maximum payload launch weight is 5250lbs. OOR design will be limited to 3562lbs (2762lbs dry weight/800lbs of fuel), which will allow approximately 1700lbs of design weight error margin when considering the 5200lb Titan IV (IUS) launch capacity.

The Fuel Transfer System (FTS) design should assume the use of mono-propellant (the primary fuel used in geo-synchronous satellites). With 800lbs of total fuel onboard, the OOR should also use mono-propellant to preclude the necessity of two separate fuel systems. With an anticipated total fuel load of 800lbs of mono-propellant, the transfer system is envisioned with the ability to feed both its own thrusters and the refueling system from any fuel tank/cell. This cross-feed design feature would preclude the OOR from depleting maneuvering fuel with transferable fuel still onboard and vice-versa. This would allow maximum flexibility and utilization of all onboard fuel. However, to preclude a compromise in fuel system integrity from depleting the entire onboard fuel supply, each tank should be selectively isolated from the others. Primary transfer is

envisioned through the docking mechanism; however, a secondary transfer system should be available. This secondary system might consist of remote tele-robotics using an umbilical fuel probe in the event that satellite docking is not feasible or the primary system fails.

Docking and refueling operations may preclude optimal orientation of solar arrays. Onboard battery power should be sufficient to complete a refueling operation without solar array support. Upon completion of the refueling mission, the OOR can then be reoriented to recharge batteries.

With a payload of 800lbs of fuel, in addition to the required fuel transfer system of 300lbs, the structure of the OOR must be robust. Additionally, sufficient structural integrity is required to support docking maneuvers and its associated structural stresses.

The command and control communications required by a remote/autonomous docking system are considered in addition to normal TT&C operations. Sufficient redundancy is required to ensure communications can be maintained throughout mission life.

The OOR concept is obviously not applicable to satellites currently on-orbit, and hence it must be designed with the "future customer" in mind. If the concept is to prove viable, satellites must be designed to accept fuel servicing from the OOR from initial inception. This dictates the early, standardized design of a docking and fuel transfer mechanism and specifications which will meet the needs of most if not all satellite designs. Clearance and configuration requirements for OOR docking must be identified early. From the docking port into the spacecraft would be the design responsibility of

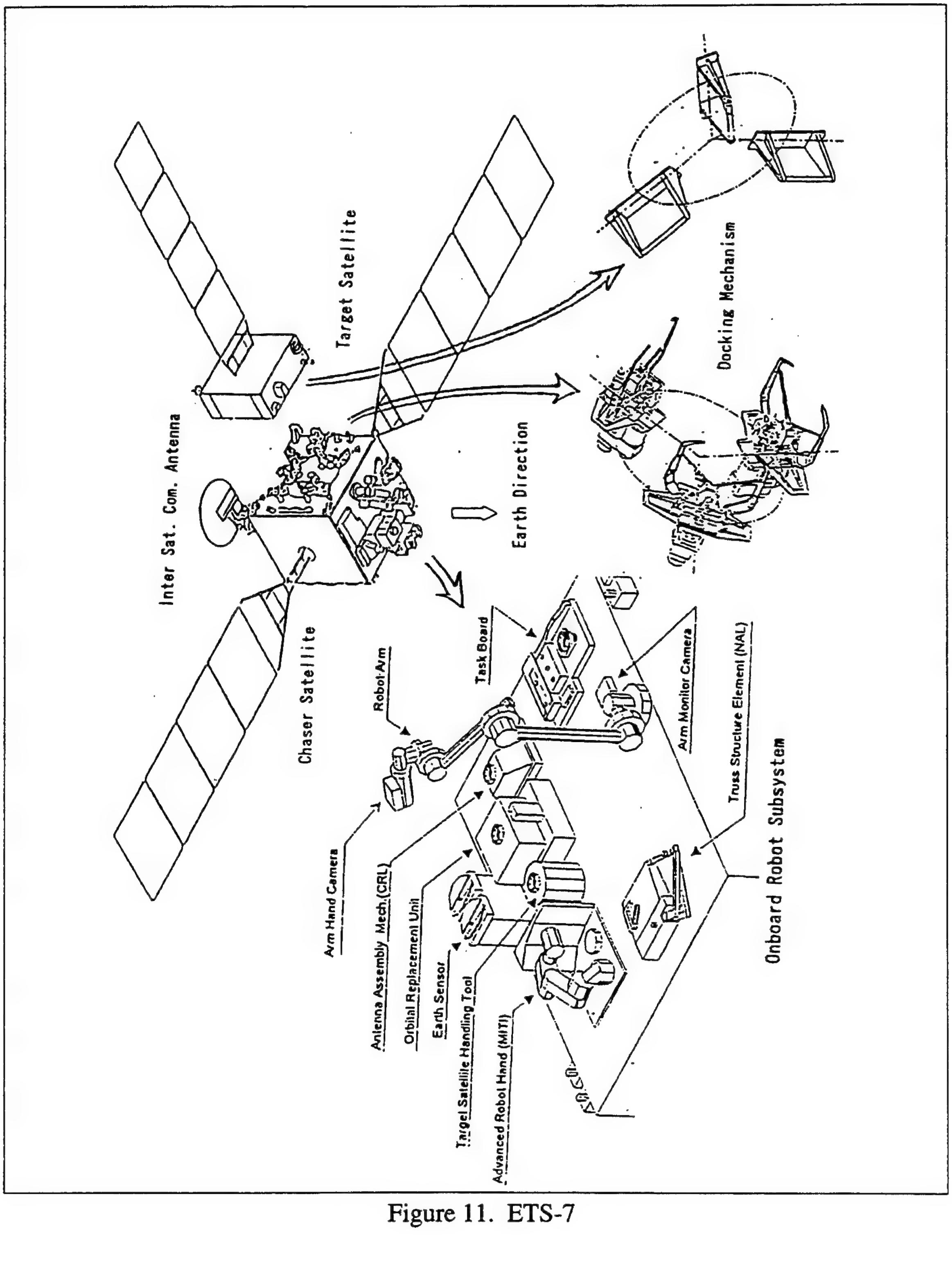


Figure 11. ETS-7

each individual satellite manufacturer, thus minimizing the concept's impact on individual satellite design. It is anticipated that through early design and integration, the docking/refueling mechanism should impose minimum impact on satellite weight budget/cost. However, later integration could prove extremely costly, depending on the design progress of the satellite. With these factors in mind a cost analysis to construct the OOR follows.

B. OOR DESIGN/COST PARAMETERS

In 1997 NASA is scheduled to fly ETS-7 to verify remote on-orbit docking and robotic repair/servicing technology. ETS-7 consists of a chaser and target satellite (Figure 11), which weigh 2.2t and 0.4t respectively. [Ref 18, p.1627] Using the approximate size/configuration of ETS-7 and DSCS-IIIB satellite costing data, a cost estimate for an on-orbit refueling vehicle was computed. Adjustments in size/ configuration for the OOR were made using the Unmanned Space System Cost Model, Seventh Edition [Ref 20] and verified using the Space Mission Analysis and Design, Second Edition [Ref 21] for an overall system "reality check" of the design process. Since specific system data and configuration is not available for ETS-7, DSCS-IIIB was chosen as a cost/equipment reference satellite due to its similar configuration to ETS-7, its geosynchronous orbit, recent technology/launch date (1994) and availability of configuration data (Appendix B) from the Unmanned Space Vehicle Cost Model. Adjustments to the OOR design from DSCS IIIB and ETS-7 are shown in Table 4-1,

along with the recommended percentage of system design parameters from SMAD. SMAD percentages are based on a historical satellite program data base.

	DSCS-IIIB	Percentage	OOR	Percentage	SMAD %
Structure	330 lbs	17.52%	750 lbs	27.50%	21.06%
Thermal	102 lbs	5.42%	165 lbs	6.05%	4.45%
Communication (DSCS Payload)	632 lbs	33.56%	307 lbs	12.9%	See Payload Below
TT&C	72 lbs	3.82%	75 lbs	2.75%	4.41%
ADCS	64 lbs	3.39%	380 lbs	11.00%	5.99%
Power	585 lbs	31.06%	585 lbs	21.45%	29.90%
Propulsion	98 lbs	5.20%	200 lbs	07.30%	4.31%
Refueling Kit (OOR Payload)	N/A	N/A	300 lbs	11.00%	28% (complies w/fuel added)
Dry Weight	1883 lbs		2762 lbs		

Table 4-1

NOTE: - Communications weight was adjusted downward from DSCS-IIIB, as communications is the DSCS primary mission.

- Propulsion weight was adjusted upward from DSCS- IIIB, as docking maneuvers will require a more precise system/propulsion between satellite refuels.

- Structure weight was adjusted upward, necessary to support the added weight of fuel to the refueling system.

- ADCS was adjusted upward to include three axis stabilization, using momentum wheels.

C. CALCULATIONS FOR RECURRING COSTS

[All Equations from Unmanned Space Vehicle Cost Model, Seventh Edition (1994) Ref 20]

The Unmanned Space Vehicle Cost Model (USCM) is published by the U.S. Air Force Material Command, Space and Missile Systems Center, Los Angeles, California. This manual, which was used to estimate the OOR cost, is a parametric estimating tool based on cost estimating relationships (CER) built from a factual historical database. Satellite systems have been broken down into costing factors/equations which can be used for cost estimates for future satellite systems. The USCM breaks satellite program costs into six basic satellite subsystems: Power, Structure, Attitude Determination and Control (ADCS), Tracking, Telemetry and Control (TT&C), Propulsion, and Thermal Control. For each of these subsystems, the USCM shows the primary costing factors associated with it based on a satellite design/cost historical data base. Additionally, the payload must be addressed for cost analysis. On DSCS-IIIB, the communications system was the payload; however, on the OOR the refueling package would be considered the payload. The cost analysis for the OOR is computed below. Specific cost multiples or cost estimating relationships (CERs) have been derived from previous satellite programs. These CERs can be applied to a specific example to estimate system cost, typically using system or component weight. All cost values computed are in thousands of dollars and recurring cost estimates are summarized in Table 4-2.

1. STRUCTURE

Spacecraft Structure	*750 lbs*

$Y=(5.838)(X1)$

Where $X1=$ Structure Weight

$Y=$ CER value for Spacecraft Structure

Therefore $Y= 4378.5$

2. THERMAL

Thermal Weight	
- Active Thermal Weight	*71 lbs*
- Passive Thermal Weight	*94 lbs*
- Total Thermal Weight	*165 lbs*

$Y=76.171 + (12.187)(X1) + (4.511)(X2)$

Where $X1=$ Active Weight

$X2=$ Passive Weight

$Y=$ CER value for Spacecraft Structure

Therefore $Y= 1365.48$

3. ADCS

ADCS	
- Attitude Determination Suite Weight	*180 lbs*
- RCS Suite Weight	*200 lbs*
- Total ADCS Weight	*380 lbs*

$Y=(250.542)(X1^{0.735})$

Where $X1 =$ Attitude Determination Suite Weight

$Y =$ CER value for ADCS (Attitude Determination)

Therefore $Y = 11{,}389.53$

$Y=(27.667)(X1^{0.619})(X2^{0.473})$

Where X1 = Reaction Control System Suite Weight

X2 = Design Life (10/2 Yrs)

Y = CER value for ADCS(Reaction Control)

Therefore Y = 2184.24 - 10 yrs/1020 - 2 years

4. ELECTRICAL POWER SYSTEM

EPS	
- Number of Solar Cells	*3000*
- Generation Suite Weight	*32*

$Y=(7.894)(X1^{0.588})$

Where X1= Number of Solar Cells

Y = CER value for Power Generation

Therefore Y = 874.67

EPS	
- Beginning of Life Power	*1200 Watts*
- Storage Suite Weight	*135 lbs*

$Y=(2.722)(X1^{0.848})$

Where X1= Beginning of Life Power

Y = CER value for Power Storage

Therefore Y = 1111.82

EPS Suite Weight	*585 lbs*

$Y=(58.775)(X1^{0.713})$

Where X1= PCD Suite Weight

Y = CER value for Power Conditioning and Distribution

Therefore **$Y = 5523.0$**

5. TELEMETRY, TRACKING AND CONTROL

TT&C Transmitter (2)	
- UHF	*4 lbs*
- SHF	*6 lbs*

$Y=(76.928) + (20.435)(X1)$

Where X1= Transmitter Weight

Y = CER value for TT&C Transmitter

Therefore **$Y = 158.67$ (UHF)**
$Y = 199.54$ (SHF)

TT&C Receiver/Exciter	*9 lbs*

$Y=(47.359)(X1^{1.105})(X2^{0.420})$

Where X1= Receive/Exciter Suite Weight

X2= Number of Receiver Boxes

Y = CER value for TT&C Receiver/Exciter

Therefore **$Y = 718.25$**

TT&C Transponder (2)	*18 lbs*

$Y=(377.529)(X1^{0.281})$

Where X1= Transponder Weight

Y = CER value for Power Storage

Therefore **$Y = 850.52$**

TT&C Digital Electronics	
- Suite Weight	*23 lbs*
- Number of Digital Elect Boxes	*5*
- Number of Links	*2*

$Y=(23.406)(X1^{0.922})(X2^{0.659})(X3^{1.091})$

Where $X1$= Digital Electronics Suite Weight

$X2$= Number of Digital Electronic Boxes

$X3$= Number of Links

Y = CER value for TT&C Digital Electronics

Therefore $Y = 2593.49$

TT&C Analog Electronics	
- Suite Weight	*1.2 lbs*
- Solenoid Driver (4)	*2.5 lbs*
- Squib Driver (4)	*2.5 lbs*

$Y=(113.777)(X1^{0.519})$

Where $X1$= Analog Electronics Weight

Y = CER value for TT&C Analog Electronic

Solenoid Driver (qty) $Y1= Y(qty^{0.926})$
Squib Driver (qty) $Y = (13.777)(X2^{0.519})(Qty^{.926})$

Where $X2$= Squib Driver Weight

Solenoid= 451.5 **Squib= 660.83**

TT&C Antenna(Horn & Radiator)	
- Horn & Radiator	*4 lbs*
- Gain	*.3 db/10*
- Wavelength	*.5 ft*
- Effective Area	*.5 sqft*

$Y=(119.351)(X1^{0.708})(X2^{0.240})$

Where $X1$= Antenna Weight

$X2$= Effective Area

Y = CER value for TT&C Antenna (Horn & Radiator)

Therefore $Y = 269.67$

TT&C Antenna (Dipoles)	*1 lb*

$Y=(26.609)(X1^{1.070})$

Where $X1$= Antenna Weight

Y = CER value for TT&C Antenna (Dipoles)

Therefore $Y = 26.61$

TT&C Antenna (S-Band)	
- S-Band Weight	*.67 lbs*
- Gain	*.26 db/10*
- Wavelength	*.5 ft*
- Effective Area	*.45 sqft*

$Y=(64.560)(X1^{1.009})(X2^{0.315})$

Where $X1$= Antenna Weight

$X2$= Effective Area

Y = CER value for TT&C Antenna (S-Band)

Therefore $Y = 33.51$

TT&C RF Distribution	*2.4 lb*

$Y=(-7.386) + (29.180)(X1) + (70.676)(X2)$

Where $X1$= RF Distribution Weight

$X2$= Active (1 = Yes, 0 = No)

Y = CER value for TT&C RF Distribution

Therefore $Y = 133.32$

6. COMMUNICATIONS

Communications Transmitter (TWTA)	
- TWTA Weight	*14.6 lbs*
- Output Power	*25 Watts*
- Frequency	*2.1 Ghz*

- WPF	*23*

$Y=(22.196)(X1^{0.727})(X2^{0.280})$

Where X1= TWTA Weight

X2= WPF - Weighted Composite Variable

Y = CER value for Communications Transmitter (TWTA)

Therefore Y = 375.0

Communications Transmitter (Solid State)	
- Solid State Transmitter Weight	*51.26 lbs*
- Output Power	*40 Watts*
- Component Quantity	*1*

$Y=(338.550) + (25.557)(X1) + (9.985)(X2)$

Where X1= Solid State Transmitter Weight

X2= Output Power

Y = CER value for Communications Transmitter (Solid State)

Therefore Y = 2048.0

Communications Receiver/Exciter Weight (2)	*30 lbs*

$Y=(193.30)(X1^{0.675})$

Where X1= Receiver/Exciter Suite Weight

Y = CER value for Communications Receiver/Exciter

Therefore Y = 1920.0

Communications Transponder Weight (2)	*30 lbs*

$Y=(67.433)(X1)$

Where X1= Transponder Weight

Y = CER value for Communications Transponder

Therefore **Y = 2023.0**

Communications Digital Electronics Weight	*56.96 lbs*

$Y=(515.079)(X1^{0.379})$

Where X1= Digital Electronics Suite Weight

Y = CER value for Communications Digital Electronics

Therefore **Y = 2383.61**

Communications	
- Weight of Other Antenna Components	*10 lbs*
- Weight of Horn, Dish	*5 lbs*
- Antenna Suite Weight	*141 lbs*

$Y=(35.473)(X1) + (24.835)(X2)$

Where X1= Weight of Other Antenna Components

X2= Weight of Horn, UHF Dish

Y = CER value for Communications Antenna

Therefore **Y = 727.26**

Communications Antenna (Reflectors)	
- Antenna Reflector Diameter Squared	*8 sqft*

$Y=(75.849)(X1^{0.935})$

Where X1= Antenna Reflector Diameter Squared

Y = CER value for Communications Antenna Reflectors

Therefore **Y = 530.07**

Communications RF Distribution	
- RF Distribution Suite Active Weight	*6 lbs*
- RF Distribution Suite Wave Guide Weight	*6 lbs*

$Y=(82.601)(X1) + (11.856)(X2)$

Where $X1$= Distribution Suite Active Weight

$X2$= RF Distribution Suite Wave Guide Weight

Y = CER value for Communications RF Distribution

Therefore Y = 566.74

7. FUEL TRANSFER SYSTEM (DRY WT)

FTS Total Weight	*300 lbs*

$Y=3(X^{1.22})$

Where X = FTS Total Weight

Y = CER value for FTS

Therefore Y = 3156.52

8. INTEGRATION ASSEMBLY AND TEST (IA&T)

IA&T	
- Spacecraft Weight	*2462 lbs*
- Fuel Transfer System Total Weight (Payload)	*300 lbs*
- Weight	*2762 lbs*

$Y=(4.833)(X1)$

Where $X1$= Spacecraft Weight + Payload Total Weight

Y = CER value for IA&T

Therefore Y = 13,348.75

9. PROGRAM LEVEL

Spacecraft Vehicle Total Recurring Cost	*67197.69*

$Y=(.289)(X1)$

Where $X1$= Spacecraft Total Recurring Cost

Y = CER value for Program Level

Therefore **Y = 19420.13**

10. LOOS - (3 AXIS STABILIZED SATELLITES)

Spacecraft Weight	*2762 lbs*

Y=(2.212)(X1)

Where X1= Spacecraft Weight + Payload Total Weight

Y = CER value for Operations and Orbital Support

Therefore **Y = 6695.72**

SMAD cost modeling calls for a cost adjustment (multiplier) greater than 1.1, for satellite designs employing new technology or advanced development concepts. This cost multiple deals with the uncertainty of new technology and the associated integration issues. For the purpose of this cost analysis a cost adjustment multiple of 1.3 will be used due to the incorporation of remote docking using visual references and the fuel transfer system in the OOR. This yields a final cost estimate of:

($86617) x (1.3) = $112.602 rounded to **113 Million**

This figure shall be used as OOR recurring costs for the cost effectiveness calculations in Chapter V. Specific non-recurring cost data for future satellite programs is not known. Although difficult to accurately estimate, an attempt to predict OOR non-recurring costs is summarized in Appendix C. However, it will be assumed that many of these cost would be offset by similar costs in repalcement satellite programs.

RECURRING COST SUMMARY
(in Thousands of Dollars)

Structure	4378.50
Thermal	1365.48
Attitude Determination & Control	11389.53
ADCS - Attitude Determination (RCS)	2184.24
Electrical Power System - Generation	874.67
EPS - Storage	1111.82
EPS - PCD	5523.0
Telemetry, Tracking & Command	
TT&C - Transmitter (UHF)	158.67
TT&C - Transmitter (SHF)	199.54
TT&C - Receiver/Exciter	718.12
TT&C - Transponder	850.52
TT&C - Digital Electronics	2593.49
TT&C - Analog Electronics (Solenoid)	451.50
TT&C - Analog Electronics (Squib)	660.83
TT&C - Antenna Horn & Radiator	269.67
TT&C - Antenna Dipoles	26.61
TT&C - S-Band Antennas	33.51
TT&C - RF Distribution	133.32
Communications	
Comm - Transmitter (TWTA)	375.00
Comm - Solid State	2048.00
Comm - Receiver/Exciter	1920.00
Comm - Transponder	2023.00
Comm - Digital Electronics	2383.61
Comm - Antenna	727.26
Comm - Antenna Reflectors	530.07
Comm - RF Distribution	566.74
LOOS - 3 Axis Stabilized	6695.72
Fuel Transfer System	3156.52
Total Spacecraft	53848.94
IA&T	13348.75
Program Level Cost	19420.13
Total OOR Recurring Cost	86617.82

Table 4-2

V: OOR COST FEASIBILITY ASSESSMENT

A. COST EFFECTIVENESS ANALYSIS

When evaluating the cost effectiveness of on-orbit refueling, it is tempting to use the cost/year approach which compares the replacement cost of a satellite and its design life to the OOR cost and its capability to refuel some specific number of satellites. This approach would yield:

	Cost	Life or Life Delta	Cost/Year of Satellite Life
Replacement Satellite	$98.85M	11.27 yrs	$8.77M
OOR Vehicle	$113M	i Satellite(s) serviced (varies 1-N) with a life delta of 3yrs	
		N = 1	$37.67M
		N = 2	$18.83M
		N = 3	$12.55M
		N = 4	$9.42M
		N = 5	$7.53M

Table 5-1

This comparison results in an apparent break-even point for refueling/replacement at 5 satellites serviced versus one replacement. However, this comparison does not take into account launch cost of either the replacement satellite or the OOR (the OOR is the same approximate weight class as the average satellite launched since 1990). When launch costs are considered the results appear slightly different. Specifically, with each satellite

refueled on-orbit the associated launch cost of the necessary replacement and (hence, not-launched) satellite is also saved. With these considerations in mind, we can use the following equation:

$$(R_C + L_C) \approx \sum[(S_C + L_C)_i \times (P_{L_\Delta})]$$

Where:

R_C = OOR Cost

L_C = OOR Launch Cost

P_{L_Δ} = % increase in satellite Life

S_C = Replacement Satellite Cost

L_C = Satellite Launch Cost

i = Number of Satellites Refueled/ Mission

The cost effective or break-even point of OOR can be determined when the value of the left side of this equation exceeds the value of the right

The statistical data to be used for this analysis is as follows:

EQUATION RELATED DATA	TERM	VALUE
$\bar{x}$ Satellite life delta (design life versus actual life)	% = 3.0/	3.0 years - 26%
$\bar{x}$ Satellite design	11.27	11.27 years
$\bar{x}$ Satellite cost	S_C	$98.85M
Launch cost for OOR/Satellite	L_C	$214/227M
On-Orbit Refueling Vehicle Cost	R_C	$113M

Table 5-2.

The satellite life delta of 3 years was demonstrated in Chapter III (Table 3-4), using the satellite data from Appendix A. The mean design life of satellites launched/contracted since 1990 and mean satellite cost (for satellites launched/contracted since 1990) were also derived from satellite data (See Appendix A). Satellite launch cost was determined from the International Reference Guide To Space Launch Systems (1991 Edition) [Ref 19] and is shown in Table 5-3. The Titan launch platform with IUS was chosen for its ability to place the OOR/satellites in geosynchronous orbit.

LAUNCH VEHICLE	MAX PAYLOAD TO GEOSYNCHRONOUS ORBIT	COST
Titan IV (IUS)	5250 lbs (2380 kg)	$214M
Titan IV (Centaur)	10000 lbs (4540 kg)	$227M

Table 5-3

Utilizing the values from Table 5-2 ($214M launch cost), the cost analysis looks like this:

$$(R_C + L_C) = \sum[(S_C + L_C)_i \times (P_{L\Delta})] \qquad \text{(Where i varies from 1 to N)}$$

$$\begin{aligned}(\$113M + \$214M) = \sum & [(\$98.85M + \$214M)_{(i=1)}(26\%)] + \\ & [(\$98.85M + \$214M)_{(i=2)}(26\%)] + \\ & [(\$98.85M + \$214M)_{(i=3)}(26\%)] + \\ & [(\$98.85M + \$214M)_{(i=4)}(26\%)] \text{ etc.}\end{aligned}$$

The break-even or cost effective point actually falls at i = 4.02 satellites serviced.

Using the higher launch cost of $227 for a larger satellite yields a cost effective point at i = 3.85 satellites serviced.

B. SENSITIVITY ANALYSIS

A sensitivity analysis on one factor at a time yields the following results:

10% INCREASE IN:	NEW VALUE	SATELLITES TO BE SERVICED
Satellite Design Life	12.4 years - 24.2%	4.32 satellites
Satellite Life Delta	3.3 years - 29.28%	3.57 satellites
Satellite Cost	$108.73M	3.90 satellites
OOR Cost	$138.6M	4.16 satellites
Launch Cost	$235.4M	4.01 satellites

Table 5-4

20% INCREASE IN:	NEW VALUE	SATELLITES TO BE SERVICED
Satellite Design Life	13.52 years - 22.18%	4.71 satellites
Satellite Life Delta	3.6 years - 31.94%	3.27 satellites
Satellite Cost	$118.62M	3.78 satellites
OOR Cost	151.2M	4.30 satellites
Launch Cost	$256.8M	4.00 satellites

Table 5-5

30% INCREASE IN:	NEW VALUE	SATELLITES TO BE SERVICED
Satellite Design Life	14.65 years - 20.47%	5.11 satellites
Satellite Life Delta	3.90 years - 34.61%	3.02 satellites
Satellite Cost	$128.51M	3.67 satellites
OOR Cost	$163.8M	4.44 satellites
Launch Cost	278.82M	3.99 satellites

Table 5-6

50% INCREASE IN:	NEW VALUE	SATELLITES TO BE SERVICED
Satellite Design Life	16.91 years - 17.74%	5.89 satellites
Satellite Life Delta	4.5 years - 39.9%	2.62 satellites
Satellite Cost	$148.28M	3.47 satellites
OOR Cost	$189M	4.72 satellites
Launch Cost	$321M	3.98 satellites

Table 5-7

Initial data analysis indicates that an increase in satellite design life would greatly impact the cost effective point of OOR, increasing the required number of satellites to be serviced to 5.89 with a 50% increase in design life. However, we are again faced with the dilemma of onboard fuel; specifically, can you carry enough to meet the increased design life? This also does not address the associated cost increase necessary to increase satellite reliability throughout design life. Increases in the satellite life delta yield promise, in that an increase from three to four years reduces the number of satellites serviced for cost effectiveness to 3.02 (Table 5-6). This is significant when you consider that the satellite life delta for the SOPER satellites (Table 3-3) is currently 3.97 years and still rising. OOR cost increases had the greatest negative impact on cost effectiveness, raising the number of necessary satellites serviced to 4.72 with a 50% cost increase. Satellite cost as well as launch cost increases were fairly insignificant.

C. RISK ASSESSMENT

Every satellite launched has an associated risk that it may not operate correctly once placed on-orbit. Extensive testing is conducted to ensure every component will operate and interface as designed. However, examples such as the Hubble telescope prove that anything is possible. The risk associated with new technology is usually higher than previously proven systems, however for the purpose of this analysis the risk of satellite/OOR failure will be considered comparable.

Launch failure risk will be considered equal in a one-to-one launch ratio. However, increased launches would represent a higher risk factor. The additional launch risk can be specifically identified through launch failure/success probability analysis. The

specific cost of risk, although not equated to a firm dollar figure, is its impact on program cost through insurance or an actual launch failure; either would greatly impact program cost. Increased satellite launches provide increased opportunity to experience a launch failure. Although insurance can mitigate launch risk, NASA, the U.S. government, and many large manufacturers typically self-insure due to excessive coverage rates (typically 10% or more of total satellite/launch costs).

D. FUEL TRANSFER

The OOR design/operations must take into account several factors which may not seem readily apparent. The first is how much fuel is available to transfer to the satellite. With an anticipated total fuel load of 800lbs of mono-propellant (primary fuel used in geosynchronous orbits), the transfer system is envisioned with the ability to cross-feed both its own thrusters and the refueling system from any fuel tank/cell. This design feature would preclude the OOR from running out of maneuvering fuel with transferable fuel still onboard and vice-versa. This would allow maximum flexibility and utilization of all onboard fuel.

The next decision is how much fuel to transfer to the satellite. The initial impulse is to "fill-it-up" as the cost difference between 100lbs and 200lbs of fuel is negligible. However, when you consider distributing the 200lbs of fuel between two separate satellites which need refueling the decision becomes more of a dilemma. Maneuvering fuel on a dead satellite is almost as worthless as a satellite with no maneuvering fuel. An option could be to conduct a statistical evaluation of remaining satellite life and fueling

for perhaps one additional year. This failure analysis would be satellite specific, requiring failure rates of onboard systems and hence will not be addressed here.

Another operation concern is where to refuel? Should there be a malfunction during refueling operations the geosynchronous slot could be filled with debris, thus making it unusable. Perhaps satellites should be boosted to a higher orbit for safety reasons. This would obviously be driven by reliability and safety design factors of the OOR as well as fuel budget. The only remaining fuel question is, how much fuel is required onboard the OOR to service the required number of cost effective satellites?

E. SATELLITE REFUELING REQUIREMENTS

[All Equations from Space Mission Analysis and Design, Second Edition Ref 21]

How much fuel must be transferred to ensure the satellite will meet the required life delta of three years? The answer to this question involves many computations based on specific satellite/orbital parameters. Satellite North/South drift as well as East/West drift compensation must be considered. Satellites in geosynchronous orbit have a N/S drift of approximately .089 degrees/year [Ref 22 p. 155]. Inclination tolerance, or how much drift above or below the equator can be tolerated, is the driving factor in fuel budget computations. Typically the time between maneuvers (Δt) is twice the time it takes the satellite to drift from the initial orbit insertion point (X_1 ,the lower limit of satellite inclination tolerance) to the equator or $\Delta t_1 + \Delta t_2$, since the drift times are equal. This is shown in Figure 12.

+.1°		----------------------------X_2--
Equator	Δt_1	--
-.1°	Δt_2	----------------------------X_1--

Figure 12. Satellite N/S Drift

To compute the time interval between N/S station keeping maneuvers (Δt):

Δt = total inclination tolerance/satellite drift

$\Delta t = (2) \times (.1)/.0897° = .2229$ yrs or 81.38 days

Total inclination tolerance is the angle the satellite must pass through to exceed tolerance. Inclination tolerance is usually payload driven, due to specific pointing or accuracy requirements. Typically the satellite is first inserted into orbit such that it is at the lower limit of drift tolerance (in this case at -.1 degrees). This position is represented by X_1,shown above in Figure 12. The satellite then drifts northward until it reaches the upper limit of tolerance (X_2), when corrective action must be taken to reposition the satellite within the payload tolerance requirements. A velocity is applied to the satellite using onboard thrusters to return the satellite to X_1. An additional velocity, equal to the first but in the opposite direction, must also be applied to stop it once it gets there. The formula for this corrective thrust or delta velocity is:

$\Delta v = 2V\sin\theta/2$ Where $\theta = 2(\Delta i)$, Δi = satellite inclination tolerance (.1 for this example)

V = Velocity in Geosychonous orbit = $(631.3481)(R_{geo})^{-1/2}$

R_{geo} = Earth radius + Satellite Altitude = 6373km + 35786km = 42164km

V = 3.075km/s

$\Delta v = 2\ (3.075)\ \sin\ (.1)$

Δv = .0107km/s (10.7m/s) which must be applied every 81.38 days (for this example)

Total N/S stationkeeping for an additional three year lifetime must take into account satellite N/S position at refueling. To believe the satellite would require fueling at the southern point of inclination tolerance would be overly optimistic. The optimum position for refueling would be at inclination of zero, or at the equator. This position would require no inclination change for the OOR, which serves to conserve onboard fuel assets. Using this positional data, the next satellite repositioning would be at half the computed Δt, since the satellite would drift north from the refueling point (equator) to the upper limit of satellite inclination and the equation above computes total satellite Δt (X_1 to X_2). In this case time to drift from the equator to X_2 would be 40 days, at which point a maneuver must be performed to move the satellite back to X_1. The normal Δt intervals of 80 days (for this example) would then apply.

Satellite mass also plays an important part in this problem. Computing the actual fuel required for the Δv, shown above, requires a formula for the mass of propellant as a fraction of initial satellite mass. Example continued:

$Mp = Mo[1-e^{-(\Delta v/(Isp \times G)}]$ Where: Δv is the computed velocity (from above)

: Isp is the specific impulse of the fuel used (typically 220-240 for mono-propellant)

: G = gravitational acceleration of the earth

For the example above: $Mp/Mo = [1-e^{-(10.7/(220 \times 9.8)}] = .004951$, which must be applied to the mass of the satellite to determine how much fuel is required.

Table 5-8 summarizes the Mp/Mo computations for some specific inclination tolerances, which will then be applied to example satellites.

Inclination Tolerance	Δt - days	Δv m/sec	Mp/Mo
.1	81.38	10.73	.004966
.3	244.15	32.20	.014824
.5	406.91	53.66	.024586
.7	569.67	75.14	.034251
1.0	813.83	107.33	.048563

Table 5-8

Applying the computed Mp/Mo for each inclination to satellites of various mass results in:

Satellite Mass	Mp/Mo .1° Incl	Fuel lbs/maneuver	Total Maneuvers/3yrs	Total Fuel Required - lbs
1500lbs	.004966	7.449278	13	105
2000lbs	.004966	9.93237	13	139
2500lbs	.004966	12.41546	13	174
3000lbs	.004966	14.89856	13	209
3500lbs	.004966	17.381	13	226

Table 5-9

Satellite Mass	Mp/Mo .3° Incl	Fuel lbs/maneuver	Total Maneuvers/3yrs	Total Fuel Required
1500lbs	.014824	22.23614	4	89
2000lbs	.014824	29.64818	4	119
2500lbs	.014824	37.06023	4	148
3000lbs	.014824	44.47227	4	178
3500lbs	.014824	51.884	4	208

Table 5-10

Satellite Mass	Mp/Mo .5° Incl	Fuel lbs/maneuver	Total Maneuvers/3yrs	Total Fuel Required
1500lbs	.024586	36.87906	3	111
2000lbs	.024586	49.17207	3	148
2500lbs	.024586	61.46509	3	185
3000lbs	.024586	73.75811	3	222
3500lbs	.024586	86.051	3	259

Table 5-11

Satellite Mass	Mp/Mo .7° Incl	Fuel lbs/maneuver	Total Maneuvers/3yrs	Total Fuel Required
1500lbs	.034251	51.37688	2	103
2000lbs	.034251	68.50251	2	137
2500lbs	.034251	85.62814	2	172
3000lbs	.034251	102.7538	2	206
3500lbs	.034251	119.8785	2	240

Table 5-12

Satellite Mass	Mp/Mo1 1.0° Incl	Fuel lbs/maneuver	Total Maneuvers/3yrs	Total Fuel Required
1500lbs	.048563	72.84478	1	73
2000lbs	.048563	97.12638	1	98
2500lbs	.048563	121.408	1	122
3000lbs	.048563	145.6896	1	146
3500lbs	.048563	169.971	1	170

Table 5-13

Longitudinal drift is primarily caused by the oblatness near the earth's equator. There are two stable positions (75 and 255 degrees East) and all satellites will drift to the closest stable point. The total Δv required to maintain longitudinal stationkeeping can be expressed as: $\Delta v = 1.735|\sin(2(L_d - L_s))|$ Where: L_d = the desired longitude

L_s = the closest stable longitude

Assuming the worst case of this equation (sin function = 1), the largest Δv possible is 1.735m/sec per year. For the assumed satellite life delta of three years, the total Δv is 5.205m/sec. Converting Δv to a percentage of satellite mass:

$Mp/Mo = [1-e^{-(\Delta v/(220 \times 9.8))}] = .00241$, which when applied to various satellites yields:

Satellite Mass	Mp/Mo	lbs Fuel Required - 3yrs
1500	.00241	3.66
2000	.00241	4.83
2500	.00241	6.02
3000	.00241	7.24
3500	.00241	8.44

Table 5-14

OOR maneuvering fuel used traveling between satellites must also be considered. Using the worst case scenario of each satellite being 180 degrees from the previous refueling target, the total velocity required to reposition (and stop) the OOR can be expressed as:

$\Delta v = 5.66(\Delta\lambda/n)$ m/sec Where: $\Delta\lambda$ = Degrees of longitude repositioning

n = Number of days required to reposition

Computing for various values of n yields:

n = X days	Δv - m/sec	Mp/Mo
30	33.96	.015628
60	16.98	.007844
90	11.32	.005237
120	8.49	.003930
180	5.66	.002622

Table 5-15

Converting this to fuel required, using Mp/Mo (the percentage of OOR mass) for each Δv computed in Table 5-15 yields:

OOR Mass	Mp/Mo	Fuel Required - lbs
3500	.015628 (n = 30days)	55
3500	.007844 (n = 60 days)	28
3500	.005237 (n = 90 days)	19
3500	.003930 (n = 120 days)	14
3500	.002622 (n = 180 days)	10

Table 5-16

It is apparent that as n continues to increase the required fuel consumption decreases. Since it is unlikely that each satellite would require servicing at the same time, planning for lower fuel consumption is viable. The time/fuel tradeoff would depend on the urgency of the refueling mission. It is cheaper to burn fuel for longitudinal changes than inclination changes. However, OOR inclination changes may be required, should a satellite be unable to be refueled at the equator. The Δv required for OOR inclination changes can be computed using the equation:

$$\Delta v = 2(Vi)(\sin\Theta/2) \quad \text{where: } Vi = \text{velocity at geosynchronous orbit (3.07kms)}$$
$$\Theta = \text{angle of orbit inclination change required}$$

Applying OOR inclination changes to various angles yields:

Θ - Inclination Angle	Δv required m/sec	Mp/Mo	OOR Mass (lbs)	Fuel Required (lbs)
1	53.58	.024735	3500	75
2	107.16	.048417	3500	146
3	160.73	.071955	3500	216
4	214.30	.094911	3500	285
5	267.82	.116889	3500	350

Table 5-17

The inclination fuel requirement is really twice the computed value shown in Table 5-17, as the OOR must be returned to an inclination of zero to service the next satellite at the optimum position. Some of this fuel cost could be mitigated by servicing the remaining satellites at the top of their inclination tolerance vice at the equator, however, it is obvious that inclination changes are not desirable due to the excessive fuel required.

OOR maneuvering fuel for docking must also be considered, although there is no specific formula for this computation. Using historical data from the Challenger rendezvous and rescue of the Solar Max satellite in 1984, 16lbs of fuel is budget for each OOR/satellite rendezvous.

VI: CONCLUSION

A. SUMMARY SYSTEM TRADES

There can be little argument that in geosynchronous orbit, fuel is the limiting factor and that the technology exists to conduct on-orbit refueling. However, the cost effectiveness of OOR is not as clear-cut to determine. Chapter IV provided the number of satellite refuelings necessary to obtain cost effectiveness. These estimates ranged from three to five satellites, using a life delta of three years. Utilizing this data coupled with the information in Chapter V on fuel consumption and budgets, an approximation of fuel required for best and worst case can be compiled. The satellite weights listed in Appendix A are satellite launch weights, fully fueled. For the purpose of this comparison, satellites will be assumed near fuel depletion and dry weight estimates will be used. Fuel budgets for geosynchronous satellites typically range from 600-800lbs of fuel. [Ref 21, p.330-332] OOR repositioning is evaluated at one less than the number of satellites to be serviced, assuming the initial OOR orbit insertion will accomplish positioning near the first satellite to be serviced. Using the basic information from Table 5-2, fuel budget estimates for the best case refueling and yields Table 6-1.

Evidenced by the data in Table 6-1, best case numbers support the refueling of seven satellites within the initial onboard fuel restriction of 800lbs. However, by evaluating a satellite inclination tolerance of .5 vice 1.0 degrees would increase fuel required for each satellite serviced by approximately 40lbs of fuel, which is shown in Table 6-2.

Satellite Mass (Dry) 1500 LBS	3 Sats Serviced Fuel Required	4 Sats Service Fuel Required	5 Sats Serviced Fuel Required	6 Sats Serviced Fuel Required	7 Sats Serviced Fuel Required
N/S Station Keeping (Incl Tolerance 1.0 degrees)	219	292	365	438	511
Satellite E/W Stationkeeping	11	15	19	23	27
OOR repositioning n = 120 days	2 Repos 28	3 Repos 32	4 Repos 46	5 Repos 60	6 Repos 74
Docking Maneuvers	(3) 48 lbs	(4) 64 lbs	(5) 80 lbs	(6) 96 lbs	(7) 110 lbs
Inclination Change	00lbs	00lbs	00lbs	00lbs	00lbs
TOTAL FUEL REQUIRED - lbs	306	407	510	617	722

Table 6-1

Note: changes in satellite life delta will alter best/worst case inclinations by changing the total N/S stationkeeping maneuvers required throughout the chosen life delta.

Satellite Mass (Dry) 1500lbs	3 Sats	4 Sats	5 Sats	6 Sats	7 Sats
TOTAL FUEL REQUIRED - lbs	426	567	710	857	1002

Table 6-2

This reduces the number of satellites able to be serviced to five, within the restriction of 800lbs of onboard fuel, which is still cost effective. Decreasing the time between satellite refuelings to n = 30 days (Table 5-16) increases each OOR repositioning fuel budget by 41lbs, which yields:

Satellite Mass (Dry) 1500lbs	3 Sats	4 Sats	5 Sats	6 Sats	7 Sats
TOTAL FUEL REQUIRED - lbs	508	690	874	1062	1248

Table 6-3

The resulting increase in OOR repositioning fuel consumption (Table 6-3) reduces the number of satellites able to be serviced to four, and hence cost effectiveness is questionable. However, with an inclination change of just one degree (Table 5-21) indicates an increase in fuel consumption of at least 75lbs, or 150lbs if you return the OOR to zero inclination. This would directly impact cost effectiveness, reducing the number of satellites able to be serviced to three.

Re-evaluating the problem using a satellite mass of 2500lbs with optimum inclination tolerance of 1 degree yields:

Satellite Mass (Dry) 2500lbs	3 Sats Serviced Fuel Required	4 Sats Serviced Fuel Required	5 Sats Serviced Fuel Required	6 Sats Serviced Fuel Required	7 Sats Serviced Fuel Required
Satellite N/S Stationkeeping (Incl Tolerance 1 degree)	366	488	610	732	854
Satellite E/W Stationkeeping	19	25	31	38	44
OOR repositioning n = 120 days	24 2 Repos	42 3 Repos	56 4 Repos	70 Repos	84 6 Repos
Docking Maneuvers	48	64	80	96	112
Inclination Change	00	00	00	00	00
TOTAL FUEL REQUIRED - lbs	461	619	777	936	1034

Table 6-4

Evidenced by the data in Table 6-4, the refueling of five satellites can be conducted within the onboard fuel restriction of 800lbs, which still meets cost effectiveness criteria.. However, by evaluating a satellite inclination tolerance of .5 degrees (worst case) fuel required for each satellite serviced would increase by approximately 65lbs, yields:

Satellite Mass (Dry) 2500lbs	3 Sats	4 Sats	5 Sats	6 Sats	7 Sats
TOTAL FUEL REQUIRED - lbs	656	879	1104	1326	1489

Table 6-5

This reduces the number of satellites able to be serviced to three, within the restriction of 800lbs of onboard fuel, which is not cost effective.

Decreasing the time between satellite refuelings to n = 30 days (Table 5-16) increases each OOR repositioning fuel budget by 41lbs, which yields:

Satellite Mass (Dry) 2500lbs	3 Sats	4 Sats	5 Sats	6 Sats	7 Sats
TOTAL FUEL REQUIRED - lbs	738	1002	1266	1531	1735

Table 6-6

The resulting increase in OOR repositioning fuel consumption (Table 6-6) does not reduce the number of satellites able to be serviced below three but cost effectiveness is certainly not going to increase.

Re-evaluating the problem using a satellite mass of 3500lbs with optimum inclination tolerance of 1 degree yields:

Satellite Mass (Dry) 3500lbs	3 Sats Serviced Fuel Required	4 Sats Serviced Fuel Required	5 Sats Serviced Fuel Required	6 Sats Serviced Fuel Required	7 Sats Serviced Fuel Required
Satellite N/S Stationkeeping (Incl Tolerance 1 degree)	510	680	850	1020	1190
Satellite E/W Stationkeeping	27	36	45	54	63
OOR repositioning n = 120 days	28 2 Repos	42 3 Repos	56 4 Repos	70 Repos	84 6 Repos
Docking Maneuvers	48	64	80	96	112
Inclination Change	00lbs	00lbs	00lbs	00lbs	00lbs
TOTAL FUEL REQUIRED - lbs	613	822	1031	1240	1449

Table 6-7

Evidenced by the data in Table 6-7, three satellites can be refueled within the 800lb OOR fuel restriction, which does not meet cost effectiveness criteria. Again evaluating a satellite inclination tolerance of .5 degrees (worst case) would increase fuel required for each satellite serviced by approximately 90lbs, yielding:

Satellite Mass 3500lbs	3 Sats	4 Sats	5 Sats	6 Sats	7 Sats
TOTAL FUEL - lbs	883	1182	1481	1780	2079

Table 6-8

This reduces the number of satellites able to be serviced to two, within the restriction of 800lbs of onboard fuel, which is not cost effective. Decreasing the time

between satellite refuelings to n = 30 days (Table 5-16) increases each OOR repositioning fuel budget by 41lbs, which yields:

Satellite Mass (Dry) 3500lbs	3 Sats	4 Sats	5 Sats	6 Sats	7 Sats
TOTAL FUEL REQUIRED - lbs	965	1305	1645	1985	2325

Table 6-9

The resulting increase in OOR repositioning fuel consumption (Table 6-9) does not reduce the number of satellites able to be serviced below two, but cost effectiveness is certainly not going to improve.

However, actual on-orbit refueling targets will probably consist of a cross section of satellite sizes, instead of all of one size as examined in the examples above. Re-evaluating the problem using a cross section of satellite sizes 1500lbs, 2500lbs, and 3500lbs with optimum inclination tolerance of 1 degree yields the results shown in Table 6-10.

Evidenced by the data in Table 6-10, five satellites can be refueled within the 800lb OOR fuel restriction, which meets cost effectiveness criteria. Again evaluating a satellite inclination tolerance of .5 degrees (worst case) would increase fuel required for each satellite serviced by approximately 40lbs, 65lbs, and 90lbs, respectively, and is shown in Table 6-11.

Satellite Mass (Dry) 2 of each rotation order - 1500, 2500, & 3500lbs	3 Sats Serviced Fuel Required	4 Sats Serviced Fuel Required	5 Sats Serviced Fuel Required	6 Sats Serviced Fuel Required	7 Sats Serviced Fuel Required
Satellite N/S Stationkeeping (Incl Tolerance 1 degree)	365	438	560	682	982
Satellite E/W Stationkeeping	20	40	52	61	65
OOR repositioning n = 120 days	28 2 Repos	42 3 Repos	56 4 Repos	70 Repos	84 6 Repos
Docking Maneuvers	48	64	80	96	112
Inclination Change	00lbs	00lbs	00lbs	00lbs	00lbs
TOTAL FUEL REQUIRED - lbs	461	584	748	909	1243

Table 6-10

Satellite Mass 3500lbs	3 Sats	4 Sats	5 Sats	6 Sats	7 Sats
TOTAL FUEL - lbs	656	819	1048	1299	1673

Table 6-11

This reduces the number of satellites able to be serviced to three, within the restriction of 800lbs of onboard fuel, which is not cost effective. Decreasing the time between satellite refuelings to n = 30 days (Table 5-16) increases each OOR repositioning fuel budget by 41lbs, which yields:

Satellite Mass (Dry) 3500lbs	3 Sats	4 Sats	5 Sats	6 Sats	7 Sats
TOTAL FUEL REQUIRED - lbs	738	942	1221	1504	1919

Table 6-12

The resulting increase in OOR repositioning fuel consumption (Table 6-12) does not reduce the number of satellites able to be serviced below three, but cost effectiveness is certainly not going to improve.

Obviously, as satellite mass increases the cost effectiveness of on-orbit refueling decreases. However, the initial design limitation of 800lbs is not carved in stone. With an increase to 1250lbs of fuel, the cost effectiveness for five satellites can be maintained throughout all examples with the exception of 3500lb satellites computed in Tables 6-7/8/9. Launch capability of the Delta IV-IUS is 5200lbs which would allow for an increased fuel payload. Consulting the Appendix A satellite data reveals that only 20 satellites exceed 3000lbs fueled, hence limiting the possibility of the latter fuel computational restrictions shown in Tables 6-7/8/9. By increasing the OOR fuel payload, the impact on the OOR cost would be minimal, with OOR structure and the fuel transfer package being the most obvious areas for cost increases (i.e., a larger/heavier structure in order to support the additional fuel weight and a heavier fuel transfer package for the additional fuel tanks and piping required). These OOR cost areas (discussed in Chapter IV) do not carry a significant cost multiple and hence would not greatly impact OOR cost. Additionally, factors such as time between satellite refueling and inclination changes can be managed to reduce fuel impact.

B. SOME FINAL THOUGHTS

Planning and cost analysis was done assuming the OOR was a "throw away" or one time only use vehicle. If the OOR was constructed using modular/ORU fuel cells and could be refueled in space for additional missions, this would greatly improve OOR cost effectiveness. This "refueling of the refueler" would probably have to occur in low earth orbit (LEO), perhaps as a space station mission. Replacement fuel cells could be launched onboard shuttle flights as space available cargo, thus saving launch costs for future OOR missions and further enhancing cost effectiveness. However, the de-orbit to LEO would have negative impact on OOR fuel. This impact could possibly be limited by the Reusable Orbital Transfer Vehicle concept which proposes use of the Earth's atmosphere to slow and capture the spacecraft, thus obtaining low earth orbit after initial de-orbit. This concept would require further analysis which exceeds the scope of this paper.

Satellite on-orbit refueling is both cost effective and tactically significant. As satellite program costs continue to increase and operations and research budgets continue to decrease the cost savings and operational flexibility provided by on-orbit refueling cannot be ignored.

APPENDIX A:

SATELLITE DATA SUMMARY
[Ref 7 and 22]

SATELLITE	LAUNCHED	LAUNCH WEIGHT KG	SATELLITE COST $M	LAUNCH VEHICLE	DESIGN vs ACTUAL LIFE	REMARKS
COMSTAR D1	05/13/76	1516	21.25	ATLAS CENTAUR	7\8.3	FUEL DEPLETED 1984
COMSTAR D3	05/29/76	1516	21.25	ATLAS CENTAUR	7\10	FUEL DEPLETED 1986
COMSTAR D2	07/22/76	1516	21.25	ATLAS CENTAUR	7\17	FUEL DEPLETED 1993
FLTSATCOM 1	02/09/78	1005		ATLAS CENTAUR	5\18 SOPER	FUEL CONSERVE OPS(FCO) MAR 95
FLTSATCOM 2	05/04/79	1005		ATLAS CENTAUR	5\15	FCO EOL 1992
WESTAR 3	08/10/79	574		DELTA	7\11	EOL 1990
FLTSATCOM 3	01/17/80	1005		ATLAS CENTAUR	5\15	FCO
GOES 4	09/09/80	900	25.66		5\2 FAILED	EOL 1988 SUPERSYNC BOOST
FLTSATCOM 4	10/30/80	1005		ATLAS CENTAUR	5\15.3	FCO
SBS 1	11/15/80	546	50	DELTA	7\9.4	FUEL DEPLETED 1990
INTELSAT 502	12/06/80	1928	33	ATLAS CENTAUR	7\16 SOPER	FCO DEC 1988
COMSTAR D4	02/21/81	1516	21.25	ATLAS CENTAUR	7\15 SOPER	
GOES 5	05/22/81	900		THOR DELTA	5\3-8	MAGE FAIL '84>RELAY MSN, FUEL '89
INTELSAT 501	05/23/81	1928	33	ATLAS CENTAUR	7\15 SOPER	FCO MAY 1988
FLTSATCOM 5	08/06/81	1005		ATLAS CENTAUR	5\5	DAMAGED ON LAUNCH EOL 1986
SBS 2	09/24/81	546	50	DELTA	7\14.7 SOPER	FCO OCT 1989
SATCOM 3R	11/20/81	1078		DELTA	10\10	EOL 1991
INTELSAT 503	12/15/81	1928	33	ATLAS CENTAUR	7\15 SOPER	FCO MAY 1989
SBS 3	01/11/82	546	50	SHUTTLE	7\13.5	FUEL DEPLETED JUN 1995
SATCOM 4	01/15/82	1078		DELTA	10\9	EOL 1991
WESTAR 4	02/26/82	572	50	DELTA	10\9.5	FAILED NOV 1991
INTELSAT 504	03/05/82	1928	33	ATLAS CENTAUR	7\13.5	FCO APR 1989, DIED NOV 1995
WESTAR 5	06/08/82	1100	50	DELTA	10\10	FUEL DEPLETED MAY 1992
INTELSAT 505	09/28/82	1928	33	ATLAS CENTAUR	7\14 SOPER	FCO APR 1989
AURORA 1	10/28/82	1084		THOR DELTA	10\14 SOPER	OK
DSCS III-1	10/30/82	1042	150	TITAN	10\14	FCO MAR 1995
DSCS II-15	10/30/82	590		TITAN	10\14	FCO MAR 1995
TDRS F1	04/04/83	2200	100	SHUTTLE	10\12 SOPER	US FAIL-REQ USE OF 370KG MAN FUE
SATCOM 1R	04/11/83	1120		DELTA	10\	EOL
GOES 6	04/26/83	900	50	THOR DELTA	5\6-9	IMAG FAIL '89\COMM RELAY '92 FUEL

SATELLITE	LAUNCHED	LAUNCH WEIGHT KG	SATELLITE COST $M	LAUNCH VEHICLE	DESIGN vs ACTUAL LIFE	REMARKS
INTELSAT 506	05/19/83	1928	33	ATLAS CENTAUR	7\13 SOPER	FCO JAN 1991
GALAXY 1	06/28/83	1200	33.33	DELTA	10\SOPER3	
TELESTAR 3A	07/29/83	3423	45.66	DELTA	10\13 SOPER	FCO SEP 1993
SATCOM 2R	09/08/83	1120		DELTA	10\11.5	EOL 1994
GALAXY 2	09/22/83	1200	33.33	DELTA	10\10.5	FCO
INTELSAT 508	03/05/84	1928	33	ARIANE	9\11	EOL 1995
SPACENET 1	05/02/84	1195	75	ARIANE	10\12 SOPER	CHINASAT FCO
INTELSAT 510	06/09/84	2013		ATLAS CENTAUR	9\12 SOPER	FCO AUG 1992
TELESTAR 3C	08/30/84	3423	45.66	SHUTTLE	10\12 SOPER	FCO JAN 1995
LEASAT 2	08/30/84	6894	85	SHUTTLE	7\11.5	FUEL DEPLETED MAY 1994
SBS 4	08/30/84	1117	50	SHUTTLE	10\11.5	FCO
GALAXY 3	09/21/84	1200	33.33	DELTA	10\11	FUEL DEPLETED OCT 1995
LEASAT 1	11/08/84	6894	85	SHUTTLE	7\8	FAILED SEP 1992
WESTAR 6	02/04/84		75			BOOST FAIL REQD REPAIR
SPACENET 2	11/10/84	1195	75	ARIANE	10\12 SOPER	OK
LEASAT 3	04/12/85	6894	85	SHUTTLE	7\10.3 SOPER	OK
GSTAR 1	05/08/85	1200	33.33	ARIANE	10\11 SOPER	FCO
TELESTAR 3D	06/17/85	3423	45.66	SHUTTLE	10\11 SOPER	FCO
INTELSAT 511	06/29/85	2013		ATLAS CENTAUR	9\11 SOPER	FCO AUG 1994
ASC 1	08/27/85	1250	33.33	SHUTTLE	10\9	EOL NOV 1994?FCO
INTELSAT 512	09/28/85	2013		ATLAS CENTAUR	9\11 SOPER	FCO AUG 1994
DSCS III-2	10/03/85	1170	150	SHUTTLE	10\11 SOPER	
DSCS III-3	10/03/85	1170	150	SHUTTLE	10\11 SOPER	
INTELSAT 507	10/19/85	1928	33	ARIANE	7\11 SOPER	FCO AUG 1990
SATCOM K2	11/27/85	1900	38	SHUTTLE	10\11 SOPER	OK
SATCOM K1	01/12/86	1900	38	SHUTTLE	10\10 SOPER	OK
GSTAR 2	03/28/86	1270	33.33	ARIANE	10\10 SOPER	FCO OCT 1993
FLTSATCOM 7	12/04/86	1100	125	ATLAS CENTAUR	5\9.1 SOPER	FCO
GOES 7	02/26/87	835	92	DELTA	5\8	FUEL DUE TO NUM OPMOVES 1995

SATELLITE	LAUNCHED	LAUNCH WEIGHT KG	SATELLITE COST $M	LAUNCH VEHICLE	DESIGN vs ACTUAL LIFE	REMARKS
SATELLITES PAST THIS POINT HAVE NOT REACHED DESIGN LIFE, BUT ARE STILL OPERATIONAL, INCLUDED FOR COST ANALYSIS ETC..						
SPACENET 3R	03/11/88	1250	75	ARIANE	10\8 SOPER	OK
INTELSAT 513	05/17/88	2013		ARIANE	9\8 SOPER	FCO AUG 1995
PAS 1	06/15/88	1220	40	ARIANE 4	11\8 SOPER	FUEL FOR 13.5 YEARS
GSTAR 3	09/08/88	1250	33.33	ARIANE	10\8 SOPER	BOOST FAIL- USED 80% MAN FUEL
SBS 5	09/08/88	1241	50	ARIANE	10\8 SOPER	OK
TDRS F3	09/29/88	2200	250	SHUTTLE	10\8 SOPER	FCO
INTELSAT 515	01/27/89	2013		ARIANE	9\7 SOPER	FCO MAY 1997
TDRS F4	03/13/89	2200	100	SHUTTLE	10\8 SOPER	
DSCS III-4	09/04/89	1170	150	TITAN	10\6 SOPER	
FLTSATCOM 8	09/25/89	1100		ATLAS CENTAUR	7\6.3 SOPER	
INTELSAT 602	10/27/89	4600	140	ARIANE	13\7 SOPER	OK
SATELLITES LAUNCHED SINCE 1990						
LEASAT 5		6894	85	SHUTTLE	7\6 SOPER	OK
INTELSAT 603		4600		TITAN	13\6 SOPER	NEW KICK MTR INSTALL STS49 1992
INTELSAT 604		4600	140	TITAN	13\6 SOPER	OK
GALAXY 6	01/09/90	1212		ARIANE 4	10\6 SOPER	OK
SBS 6	03/14/90	2478	60	ARIANE 4	10\6 SOPER	ONBOARD FUEL FOR 15.6 YEARS
SATCOM C1	06/23/90	1169		ARIANE	10\6 SOPER	OK
GSTAR 4	10/12/90	1295		ARIANE	10\6 SOPER	
SPACENET 4	10/12/90	728	75	DELTA 2	10\5 SOPER	OK
AURORA 2	11/20/90	1338	88	DELTA	12\5 SOPER	
TDRS F5	11/20/90	2200	30	SHUTTLE	10\5 SOPER	OK
INTELSAT 605	04/13/91	4600	140	ARIANE	15\6 SOPER	OK
INTELSAT 601	05/29/91	4600	140	ARIANE	13\5 SOPER	OK
GALAXY 5	08/02/91	1412		ATLAS 1	10\4 SOPER	ONLY FUEL FOR 9 YEARS
GALAXY 7	08/14/91	2986		ARIANE 4	15\4 SOPER	ONLY FUEL FOR 12 YEARS
SATCOM C4	10/29/91	1402		DELTA	12\4 SOPER	OK
SATCOM C3	03/14/92	1375		ARIANE 4	12\4 SOPER	

SATELLITE	LAUNCHED	LAUNCH WEIGHT KG	SATELLITE COST $M	LAUNCH VEHICLE	DESIGN vs ACTUAL LIFE	REMARKS
INTELSAT K	08/01/92	2836	95	ATLAS CENTAUR	10\6 SOPER	OK
INTELSAT 701	08/31/92	3610	78.85	ARIANE 44	65% at 10yrs	
INTELSAT 702	09/10/92	3610	78.85	ARIANE 44	65% at 10yrs	
INTELSAT 703	01/13/93	3610	78.85	ATLAS II	65% at 10yrs	
INTELSAT 704	03/25/93	3610	78.85	ATLAS II	65% at 10yrs	
INTELSAT 705	09/03/93	3610	78.85	ATLAS II	65% at 10yrs	
INTELSAT 706	10/22/93	4643		ARIANE 44	65% at 10yrs	FUEL FOR 15YRS
INTELSAT 707	06/17/94	4643		LONG MARCH3B	65% at 10yrs	LAUNCH FAILURE
INTELSAT 708	06/24/94	4643	140	ARIANE 44	65% at 10yrs	
INTELSAT 709	01/01/95	4643	90	ARIANE 44	65% at 10yrs	
PAS 4	03/01/95	2985	75		11	
PAS 3R	03/08/95	2985	75	ATLAS IIA	10	
GALAXY 3R	05/01/95	3380			10	
GALAXY 9	05/31/95					
TDRS 6	09/01/95	2200		SHUTTLE	10	
TDRS 7	10/01/95	2200		SHUTTLE	10	
TELSTAR 402R	10/22/95			ARIANE	12	
UHF 1 (UFO)	12/15/95	3000	172.2	ATLAS CENTAUR	10-15YRS	PARTIAL LAUNCH FAILURE - ORBIT
UHF 2	01/01/96	3000	64.4	ATLAS CENTAUR	10-15YRS	
UHF 3	03/01/96	3000	60.1	ATLAS CENTAUR	10-15YRS	
UHF 4	05/24/96	3000	87.2	ATLAS CENTAUR	10-15YRS	
UHF 5	06/01/96	3000	79.3	ATLAS CENTAUR	10-15YRS	
UHF 6	12/01/96	3000	65.4	ATLAS CENTAUR	10-15YRS	
LAUNCHED MEAN		**3083.7027027**	**89.8270833333**			
CONTRACTS						**FOR COST ANALYSIS ONLY**
AMSC/MSAT	12/01/90		100			

SATELLITE	LAUNCHED	LAUNCH WEIGHT KG	SATELLITE COST $M	LAUNCH VEHICLE	DESIGN vs ACTUAL LIFE	REMARKS
INMARSAT F1	02/01/91		80			
INMARSAT F2	02/01/91		80			
INMARSAT F3	02/01/91		80			
INMARSAT F4	02/01/91		80			
ASIASAT (GE)	09/01/92		133			
INTELSAT 801	09/01/92		82.5			
INTELSAT 802	09/01/92		82.5			
INTELSAT 806	03/01/94		82.3			
TDRS 8	02/01/96		160.5			
TDRS 9	02/01/96		160.5			
TDRS 10	02/01/96		160.5			
UHF 7		3000	94.4		10-15YRS	
UHF 8		3000	121		10-15YRS	
UHF 9		3000	166		10-15YRS	
UHF 10		3000	135		10-15YRS	
CONTRACT MEAN			**112.3875**			

LAUNCHED/CONTRACT MEAN FOR SATELLITES SINCE 1990	**98.85125**

APPENDIX B:

DSCS IIIB SATELLITE DATA

[Ref 20]

SATELLITE NAME:	DSCS IIIB

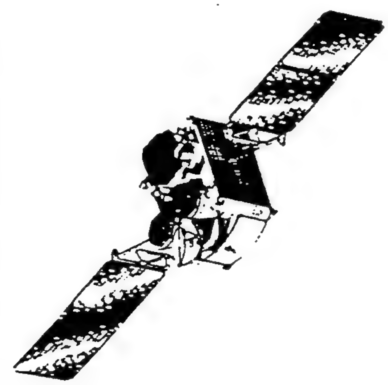
GENERAL

BLOCK & DESIGNATED UNITS:	Block B, Units B4 through B7
USER/CONTRACTING:	AF SMC
CONTRACTOR:	General Electric
CONTRACT AWARD:	15 January 1982
CONTRACT COMPLETION:	
TYPE OF CONTRACT:	FPIF
DOLLAR YEAR OF DATA:	1984
PREDECESSOR VEHICLES:	DSCS III — A1, A2, and A3
LAUNCH WEIGHT (LB):	
DRY WEIGHT (LB):	1883.70
LAUNCH VEHICLE:	Atlas II, Shuttle, Dual Compatible, Centaur & IUS
FIRST LAUNCH DATE:	Classified
LAST LAUNCH DATE:	May 1994
DESIGN LIFE (YR):	10
NO. OF NEW UNITS:	0
NO. OF QUAL UNITS	0
NO. OF PROD UNITS	4

ORBIT PARAMETERS

APOGEE (NMI):	PERIGEE (NMI):	INCLINATION (DEG):
Synchronous	Synchronous	0.1

SKETCH



MISSION DESCRIPTION

The DSCS III was developed for the Air Force by GE. Its mission is to provide uninterrupted secure strategic and tactical voice and data transmission, military command and control, and ground mobile communications. This is achieved by antijam abilities and high frequency wideband communications. Block B consists of 11 satellites, some of which have already been delivered. Block B satellites are covered in two data packages. Block B1 consists of satellites B4-B7 and Block B2 consists of satellites B8-B14. These satellites have some solid state amplifiers replacing TWTAs, a new X-band downlink capability for the AFSATCOM transponder, and improved security equipment.

DESCRIBE ALL CAVEATS

The average recurring cost (and resulting first unit cost) for DSCS IIIB was significantly higher than that of DSCS IIIA. The common belief is that significant overruns were incurred on the DSCS IIIA contract, particularly by the subcontractors operating under fixed price contracts. As a result, the DSCS IIIB recurring costs are more representative of the "true" recurring costs of the DSCS III program. Therefore, the data point for DSCS IIIB units 4 through 7 was used only in developing recurring CERs. Furthermore, due to the similarity of the two DSCS IIIB data points, in several cases these blocks were combined to form one data point and a new first unit cost was calculated.

MAIN SUBSYSTEM/COMPONENT COST DRIVERS

STRUCTURE	STRUCTURE WEIGHT (LB): 330.48	MATERIAL TYPE: Aluminum, magnesium, beryllium, magnesium thorium
THERMAL	THERMAL WEIGHT (LB): 101.93	AVERAGE TEMP RANGE (DEG FAHRENHEIT): 70
TYPE OF THERMAL CONTROL:	Passive/semi-passive surface coating: single and multi-layer insulation; mirrored surface; passive conduction. Active/semi-active: heaters and radiators.	

SATELLITE NAME: DSCS IIIB

MAIN SUBSYSTEM COMPONENT COST DRIVERS (Cont'd)

ATTITUDE DETERMINATION & CONTROL SYSTEM

ACS WEIGHT (LB): 162.28	NO. OF TANKS: 4	POINTING ACCURACY (DEG): 0.10

RCS WEIGHT (LB): 98.83 (included in ACS weight)

SENSOR TYPES (INCLUDE NO. OF EACH TYPE): Non-scanning Earth Sensors (2); Rate Gyro Assembly (1); Sun Sensors (2)

TANK VOLUME (CU IN): Fuel — 1388.75; Oxidizer — 4165.25

TORQUE METHODS: Reaction Wheels

ELECTRICAL POWER SYSTEM

EPS WEIGHT (LB)	BOL POWER (WATTS)	SOLAR ARRAY AREA (SQ FT)	NO. OF SOLAR CELLS	GENERATION WEIGHT (LB)
585.59	Summer solstice: 1310; Autumnal Equinox: 1397	126		31.72
BATTERY TYPE (AND NO.)	**WEIGHT OF ONE BATTERY (LB)**	**DISTRIBUTION WEIGHT (LB)**	**PCE SUITE WEIGHT (LB)**	**BATTERY CAPACITY (AMP-HR)**
NiCd (3)	45.16	211.43	113.90	35.0 per battery

APOGEE KICK MOTOR

TOTAL IMPULSE	AKM DRY WEIGHT (LB)	STABILIZATION METHOD

TELEMETRY, TRACKING, AND COMMAND SYSTEM

TT&C WEIGHT (LB)	POWER REQUIRED (WATTS)	RF POWER OUTPUT (WATTS)	TWTA OR SOLID STATE AMPS	RECEIVER WEIGHT (LB)
70.99			Solid State	9.00
RECEIVER FREQUENCY (MHz)	**TRANSMITTER WEIGHT (LB)**	**TRANSMITTER FREQUENCY (MHz)**	**TRANSPONDER WEIGHT (LB)**	**TRANSPONDER FREQUENCY (MHz)**
7600	14.16	7600	18.04	Receive A: 1807.764; B: 1823.779 Transmit A: 2257.5; B: 2277.5
DIGITAL ELECT WEIGHT (LB)	**ANALOG ELECT WEIGHT (LB)**	**ANTENNA WEIGHT (LB)**	**ANTENNA APERTURE (INCHES)**	**ANTENNA GAIN (DECIBELS)**
23.00	N/A	0.67	Not specified	65%: -7.5; 25%: -4.5

TRANSMITTER OUTPUT POWER (WATTS)	TRANSPONDER OUTPUT POWER (WATTS)	DIGITAL ELECT OUTPUT POWER (WATTS)	DATA RATE (kb/s)
0.7	2.0		Command, Real Time: 1; Telemetry, Real Time: 1

COMMUNICATION

COMM WEIGHT (LB)	POWER REQUIRED (WATTS)	RF POWER OUTPUT (WATTS)	TWTA OR SOLID STATE AMPS	RECEIVER WEIGHT (LB)
632.40			TWTA & Solid State	51.26

RECEIVER FREQUENCY (MHz): Freq Gen 5.00; SCT Converter, not specified; Freq Synth, not specified; SCT-SHF, 7975 to 8025; SCT-UHF, Classified

TRANSMITTER FREQUENCY (MHz): Freq STD, 5.0; LNA, not specified; TDAL, 7900 to 8400; TDL, not specified; TWTA—10W: 7400 to 77500; TWTA—40W: 7250 to 7400; SCT-SHF: 8000; HESSA, not specified

TRANSMITTER WT (LB)	TRANSPONDER WT (LB)	TRANSPONDER FREQUENCY (MHz)	DIGITAL ELECT WEIGHT (LB)	ANALOG ELECT WEIGHT (LB)	ANTENNA WEIGHT (LB)
131.25	N/A	N/A	56.96	9.42	305.60

ANTENNA APERTURE (INCHES): ECH-R, 6.5; ECH-T, 7.7; Gimballed Dish Antenna, 855; 61 MBA Receive, 45; 19 Transmit MBA, 28; UHF Receive, not specified; UHF Transmit, not specified

ANTENNA PEAK GAIN (DECIBELS): ECH-R, 16.8 dBi; ECH-T, 17.0 dBi; Gimballed Dish Antenna, 30.2; 61 MBA Receive, narrow coverage—29.4, earth coverage—14.4; 19 Transmit MBA, narrow coverage—26/26.5, earth coverage—s1615; UHF Receive & Transmit, classified

TRANSMITTER OUTPUT POWER (WATTS): Freq STD, 2.02; LNA, MBA—1.54, ECH—3.08; TDAL, Fetal—5.8; TDL, not specified, TWTA—10W, 10; TWTA—40W, 40; SCT-SHF, 50; HESSA, 10

TRANSPONDER OUTPUT POWER (WATTS)	DIGITAL ELECT OUTPUT POWER (WATTS)	DATA RATE (kb/s)
N/A	Not specified	SCT digital processor, real time: 0.07

APPENDIX C:

OOR NON-RECURRING COSTS ESTIMATES

[All Equations from Unmanned Space Vehicle Cost Model, Seventh Edition - Ref. 20]

Satellite non-recurring cost consists of the Research, Development, Test and Evaluation (RDT&E) which typically includes design, analysis, testing, prototypes and qualification runs. Additionally, it also includes ground station costs. The non-recurring cost estimate uses the same CER methodology used to estimate recurring cost in Chapter IV. and are summarized in Table C-1. Non-recurring cost estimates for the OOR are as follows:

1. STRUCTURE

Spacecraft Structure	*750 lbs*

$Y=(99.045)(X1)^{0.789}$

Where X1= Structure Weight

Y= CER value for Spacecraft Structure

Therefore Y= 18376.21

2. THERMAL

Thermal Weight	*165 lbs*

$Y= (0.243)(X1)^{0.597} + (X2)^{0.983}$

Where X1= Thermal Weight

X2= Satellite Weight

Y = CER value for Spacecraft Structure

Therefore Y= 12364.23

3. ADCS

ADCS	
- Determination Suite Weight	*180 lbs*
- RCS Suite Weight	*200 lbs*
- Total ADCS Weight	*380 lbs*

$Y=(666.439)(X1)^{0.711}$

Where X1= Attitude Determination Suite Weight

Y = CER value for ADCS (Attitude Determination)

Therefore Y = 26746.02

$Y=(125.998)(X1)^{0.733}$

Where X1= Reaction Control System Suite Weight

Y = CER value for ADCS(Reaction Control)

Therefore Y = 6123.77

4. ELECTRICAL POWER SYSTEM

EPS	
- Number of Solar Cells	*3000*
- Generation Suite Weight	*32 lbs*
- Beginning of Life Power	*1200 Watts*
- Storage Suite Weight	*135 lbs*
-EPS Suite Weight	*585.59*

Y=(0.025)(X1) + (0.024)(X2)

Where X1= (Generation Suite Weight)(Beginning Life Power (BOL))

X2= Number of Solar Cells

Y = CER value for Electrical Power Generation

Therefore Y = 1032

$Y=(114.127) + (2.584)(X1)$

Where $X1=$ (Weight of One Battery)(Capacity of One Battery)

$Y =$ CER value for Electrical Power Storage

Therefore $\mathbf{Y = 4183.93}$

$Y = (5.515)(X1)$

Where $X1=$ BOL Power

$Y =$ CER value for Power Conditioning and Distribution

Therefore $\mathbf{Y = 6618.0}$

5. TELEMETRY, TRACKING AND CONTROL

TT&C	
- Transmitter	*10 lbs*
- Receiver/Exciter	*9 lbs*
- Digital Electronics (2 Links)	*23 lbs*
- Antenna (4 Systems)	*4 lbs*

$Y=(67.121)(X1)$

Where $X1=$ Transmitter Suite Weight

$Y =$ CER value for TT&C Transmitter

Therefore $\mathbf{Y = 671.21}$

$Y = (-224.351) + (116.683)(X1)$

Where $X1=$ Receive/Exciter Suite Weight

$Y =$ CER value for TT&C Receiver/Exciter

Therefore $\mathbf{Y = 825.80}$

$Y = (211.243)(X1)^{0.787}(X2)^{0.853}$

Where $X1=$ Digital Electronics Suite Weight

X2= Number of Links

Y = CER value for TT&C Digital Electronics

Therefore **Y = 4500.29**

$$Y = (-222.262) + (30.670)(X1) + (480.840)(X2)$$

Where X1= Antenna Suite Weight

X2= Number of Antenna Systems

Y = CER value for TT&C Antenna

Therefore **Y = 1823.78**

6. COMMUNICATIONS

Communications Transmitter (TWTA)	
- TWTA Weight	*14.6 lbs*
- Solid State Transmitter	*51.26 lbs*
- Receiver/Exciter	*30 lbs*
- Transponder (2 units)	*30 lbs*
- Digital Electronics(5 links)	*56.96 lbs*
- Antenna (4 systems)	*141 lbs*
- Antenna Reflectors	*8 sqft*

$$Y=(524.161)(X1)^{0.875}$$

Where X1= TWTA Weight

Y = CER value for Communications Transmitter (TWTA)

Therefore **Y = 5473.61**

$$Y= (0.249)(X1)^{1.101}(X2)^{0.728}$$

Where X1= Solid State Transmitter Weight

X2= Transmitter Frequency

Y = CER value for Communications Transmitter (Solid State)

Therefore **Y= 5283.18**

$Y=(273.793)(X1)$

Where X1= Receiver/Exciter Suite Weight

Y = CER value for Communications Receiver/Exciter

Therefore **Y= 8213.79**

$Y=(682.769)(X1)^{0.463}$

Where X1= Transponder Weight

Y = CER value for Communications Transponder

Therefore **Y= 3297.47**

$Y=(211.243)(X1)^{0.787}(X2)^{0.853}$

Where X1= Digital Electronics Suite Weight

X2= Number of Links

Y = CER value for Communications Digital Electronics

Therefore **Y= 20074.15**

$Y=(-222.262) + (30.670)(X1) + (480.840)(X2)$

Where X1= Antenna Suite Weight

X2= Number of Antenna Systems

Y = CER value for Communications Antenna

Therefore **Y = 6025.57**

$Y=(1763.889)(X1)$

Where X1= Antenna Reflector Diameter Squared

Y = CER value for Communications Antenna Reflectors

Therefore **Y= 14111.11**

7. INTEGRATION ASSEMBLY AND TEST (IA&T)

IA&T	
- Spacecraft Weight	*2462 lbs*
- Fuel Transfer System (FTS) Total Weight	*300 lbs*
- Weight	*2762 lbs*

$Y= 956.384 + (0.191)(X1)$

Where X1= Spacecraft Weight + Payload (FTS) Non-Recurring Cost

Y = CER value for IA&T

Therefore **Y = 29396.41**

8. PROGRAM LEVEL

Satellite Total Recurring Cost	*113000 K*

$Y=(2.340)(X1)^{0.808}$

Where X1= Spacecraft Total Non-Recurring Cost

Y = CER value for Program Level

Therefore **Y= 28320.65**

9. AEROSPACE GROUND EQUIPMENT (AGE)

Satellite Total Non-Recurring Cost	*149976.41 K*

$Y=(8.304)(X1)^{0.638}$

Where X1= Space Vehicle Total Non-Recurring Cost

Y = CER value for Aerospace Ground Equipment

Therefore **Y = 16656.73**

NON-RECURRING COST SUMMARY
(in Thousands of Dollars)

Structure	18376.21
Thermal	12364.23
Attitude Determination & Control	26746.02
ADCS - Attitude Determination RCS	6123.77
Electrical Power Supply EPS - Generation EPS - Storage EPS - PCD	 1032.00 4183.93 6618.00
Telemetry, Tracking & Command TT&C - Transmitter TT&C - Receiver/Exciter TT&C - Digital Electronics TT&C - Antenna Suite	 671.21 825.80 4500.29 1823.78
Communications Comm - Transmitter (TWTA) Comm - Solid State Comm - Receiver/Exciter Comm - Transponder Comm - Digital Electronics Comm - Antenna Comm - Antenna Reflectors	 5473.61 5283.18 8213.79 3297.47 20074.15 6025.57 14111.11
Fuel Transfer System (EST)	3156.52
Total Spacecraft	148900.64
IA&T	29396.41
Program Level Cost	28320.65
Aerospace Ground Equipment	16656.73
Total OOR Non-Recurring Cost	223198.1

Table C-1

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